NASA TN D-40

NASA TN D-40

TECHNICAL NOTE
D-40

SOME NOTES ON ATTITUDE CONTROL OF EARTH SATELLITE VEHICLES

By Warren Gillespie, Jr., Donald G. Eide,
and Allan B. Churgin

Langley Research Center
Langley Field, Va.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

WASHINGTON December 1959

1E

CONTENTS

L
4
3
1

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TECHNICAL NOTE D-40

SOME NOTES ON ATTITUDE CONTROL OF EARTH SATELLITE VEHICLES

By Warren Gillespie, Jr., Donald G. Eide,
and Allan B. Churgin

SUMMARY

L 4 3 1

The problem of matching satellite mission stabilization requirements with attitude-control-system capabilities is investigated. The probable civilian missions in an advanced satellite program and a variety of possible satellite stabilization systems are considered. On the basis of this study it appears that most of the mission requirements can be initially met. In particular, an improved version of spin stabilization incorporating earth-horizon scan units is attractive for meteorological payloads. Attitude-control systems employing radiation-sensing units should be developed. Such systems can be quite versatile in meeting performance requirements with adequate useful life and moderate weight and power. They are also compatible with the launching vehicle guidance capability.

INTRODUCTION

The satellite missions performed during the International Geophysical Year have been generally limited to the simplest missions and to scalar measurements of the space environment. This approach has permitted the highest probability of initial success insofar as proper functioning of the payload package is concerned. With the possible exception of the third Russian satellite these vehicles have employed no complex stabilization systems for pointing the satellite or instruments in some desired direction. It is clear that such an orientation capability could broaden the scope of the mission to include directional (vector) measurements of observed phenomena as well as to increase the variety of missions that could be performed. These considerations have stimulated an overall assessment of attitude control of earth satellite vehicles. The twin objectives of this study are to determine the stabilization requirements of the probable missions to be undertaken and to indicate the type of stabilization that can most suitably meet the requirements.

SYMBOLS

A	horizon-to-horizon scan angle from satellite position
B	angle from zenith
e	ellipticity of satellite orbit
F	force
H	magnetic field intensity
h	altitude
I	inertia
I_x,I_y	moment of inertia about spin and pitch (or yaw) body axes, respectively
i	inclination of satellite orbital plane to equator
K	motor efficiency
L	length of cable required to despin satellite
l	length of cylindrical body shell
M	disturbing moment about body yaw axis
m	mass
n	regression rate of orbit nodes
p	spin rate
r_i	inside radius of cylindrical body shell
r_o	outside radius of cylindrical body shell
T	gravitational moment
t	time
t_t	time at termination of disturbing moment
V	velocity

L
4
3
1

W weight

α orientation error in plane of orbit with respect to direction of gravity

γ orbit position angle of satellite from point of injection

δ angular displacement of spinning body

ζ location angle for longitudinal element of cylindrical body shell with respect to magnetic field, in plane normal to spin axis

L431

θ,ψ,ϕ pitch, yaw, and bank angles, respectively

μ effective magnetic permeability for body shell

ξ angle of spin axis to magnetic field

σ electrical conductivity of body shell

τ time after termination of disturbing moment ($\tau = 0$ occurs at t_t)

ω circular orbit rate

Subscripts:

max maximum

n nth component

o initial

R pertaining to flywheel

S pertaining to satellite

One and two dots over symbols denote first and second time derivatives, respectively.

GENERAL CONSIDERATIONS

The mission stabilization requirements must be established and the control-system capability evaluated. The two subsystems, instrumentation package and stabilization package, must be matched to give the

overall system configuration for the satellite. Figure 1 illustrates this matching problem in block-diagram form. A most essential requirement of all missions is reliability. Successful accomplishment of the mission is the final test of reliability for that mission. References 1, 2, and 3 consider this subject in some detail. A second general requirement is performance which can vary considerably between the different missions. A certain minimum level of performance is needed in terms of pointing accuracy and response time. Some factors that affect reliability and performance of the stabilization system can be seen by examination of an information flow diagram as shown in figure 2. Everything in
the diagram affects reliability and performance. Not shown by the dia- L
gram is the additional factor of mission duration. This factor will 4
influence the type of control that may be chosen. In the event that the 3
mission duration exceeds the orbit lifetime and that the satellite or 1
its payload is to be recovered, the stabilization system will be required at initiation of reentry, and, during reentry, it may also continue to play a key role.

Other general aspects of the matching problem are the availability of launching vehicles suitable for the size of payload and orbit desired and the possibility of combining several missions or experiments. In this way a stabilization system that is undesirable for a small satellite could be quite attractive for a larger one. The same orbital conditions must be nearly optimum for all experiments so grouped, and possible physical or organizational interference between them must be considered. The cost per pound of payload can be reduced at some loss of flexibility in the program for the individual experiment.

SATELLITE MISSIONS AND STABILIZATION REQUIREMENTS

Several broad categories of civilian satellite missions are given as follows:

(1) Space environment

(2) Astronomy and astrophysics

(3) Meteorology

(4) Communications

(5) Space flight technology

This classification was arrived at from consideration of references 4 to 10.[1] It is based on what one might call the particular satellite, as for example, a meteorological satellite. Information contained in these references has been summarized and is presented in tables I to V. Items enclosed in brackets may be attributed to the present study. The orbit and stabilization requirements of some missions are not accurately known at this time. In many cases the pointing direction need only be known or the vehicle despun or slowed as for an initially spinning vehicle. The additional time for data workup, however, may well warrant the use of a more sophisticated control system.

L
4
3
1

For the space environmental missions of table I, a method of coarse attitude control could generally suit each one. Coarse control is arbitrarily defined as an orientation error greater than $\pm 1^\circ$ and fine control, an error less than $\pm 1^\circ$. Pointing toward the earth, the sky, or the sun would, in the majority of cases, probably be acceptable.

The orientation is more readily apparent for the astronomical and astrophysical missions of table II. The orientation accuracy for these missions will depend on the resolution of the data-gathering telescopes and also on whether the celestial sphere is being surveyed by a sweeping motion or whether the heavenly bodies are under steady scrutiny. Some information on the question of pointing accuracy is provided in reference 6. Use of a spectrograph would permit measurements of star spectra in great detail and require less strict orientation than telescopic images of extreme resolution. For example, a 20-inch telescope would permit, theoretically, a resolution of 0.25 second of arc and require an orientation accuracy of about 0.10 second. A larger telescope would require still more accuracy if operated at its theoretical limit of resolution. Fortunately, astronomers may be satisfied, initially, with a bit less than the theoretically possible resolution. A telescope of 20 seconds of resolution appears to be more realistic at the current stage of the satellite art. For high-resolution star tracking, the factor of "aberration of light" caused by satellite motion perpendicular to the direction of the observed star introduces a continuous tracking maneuver of about ± 5 seconds of arc for each orbital revolution of the satellite.

The meteorological mission (table III) has been considered by references 4, 5, 6, and 8 and is fairly well defined. A considerable coverage of the earth's surface is desired for observation of cloud cover as well as for radiation measurements to determine the atmospheric heat balance. An orientation accuracy of $\pm 1^\circ$ should be adequate for observations toward the earth at varying angles from the zenith.

---

[1]Also considered in this classification was information provided by R. W. Porter from Space Science Board, National Academy of Sciences, by letter to NASA.

Accurate pointing toward the sun is also essential for determination of the solar input. Measurement of incoming and outgoing radiations, by means of directional bolometers, can be accomplished in the intermittent manner proposed in reference 5, but with improved determination of the orientation. Separate vehicles might also be used to better advantage, one oriented for continuous measurement of earth-atmosphere radiations and the other for solar radiations. The atmospheric heat balance might then be determined by using the combined measurements from both vehicles. Simultaneous continuous tracking of both the earth and sun from a single satellite would be more complex and less reliable and require more time and expense for achievement.

L431

The second and third items of table IV represent difficult communications missions. References 9 and 10 consider these communications problems. In the case of satellite communications, the antenna would track the ground station by homing in on the maximum signal strength. The antenna would require separate stabilization. Accuracy of pointing must be of the order of $\pm 1/2^{\circ}$ or better. Mutual interaction or feedback between vehicle and antenna motions is an added problem. The communications satellites of worldwide communication relays have been considered in reference 10. The active relay requires less accuracy for pointing than does the passive planar reflecting type but is more restricted in what it can relay and requires power in the satellite for relaying. Pointing accuracy required for the flat reflecting type can be relaxed by using a shape intermediate to a flat and a complete spherical shape. The spherical (nonstabilized) reflector while requiring higher transmission power at ground stations represents a much easier satellite problem at the 2,000-mile orbit and should be compared with oriented relays near this altitude. Use of a single relay in a 24-hour orbit would, in addition to an accurate orientation problem, also require close altitude and overhead-location control to maintain position with respect to ground stations.

Table V presents the technological missions. This table indicates that precision orbital measurements can serve several purposes. In this case, a self-illuminated satellite would permit continuous sightings when in the earth's shadow zone irrespective of reflected sunlight from the vehicle. The light beacon should be directed toward the earth for efficient utilization of internal power for the beacon. However, the added complexity, weight, and power requirement of a stabilization system for this purpose only may not result in a significant advantage. The second item refers to the need of power-system components with regard to pointing control. Although only the solar-powered system or collector requires pointing toward the sun, all power systems require some form of cooling radiator which may be oriented from the sun for maximum cooling. The pointing accuracy for a parabolic solar collector should be about $\pm 2^{\circ}$ for maximum power generation. An oscillatory motion of varying amplitude about the sun's disk would permit control of the

power generated. This might be preferable to the use of blinds or slats for power regulation. The third, fourth, and fifth items would make use of the satellite as a facility for testing stabilization, guidance, and propulsion systems and components under true space environmental conditions and for long periods of time. Tests of this kind are needed to get the space flight program "off the ground" by determination of performance, reliability, and useful life of these systems and components. Also, under the condition of weightlessness a more accurate breed of inertial sensing and control instruments may be developed. Such components and improved systems are needed for space flight projects and in connection with reentry and recovery and manned satellites.

In summary, the required precision of orientation control is seen to vary over a broad range from coarse to extremely fine and for periods of time generally exceeding three months. Several combinations of orbit and orientation will be difficult. One such combination is an orbit of extreme eccentricity with the satellite required to point toward the earth. Another difficult combination is one that requires tracking of a fixed spot on the earth from a close orbit. A complex orientation situation occurs for a satellite in which portions of the vehicle are oriented toward the sun and the earth.

STABILIZATION METHODS AND CAPABILITIES

The types of attitude stabilization systems that are contemplated for use in more advanced earth satellite vehicles are designated in accordance with the most characteristic feature of the system and are discussed approximately in increasing order of complexity. These systems are

(1) Spin

(2) Gravitational-centrifugal-force gradient

(3) Earth-horizon scan, area scan

(4) Moon, planets, sun, and star tracking

(5) Other systems

The capabilities of each system are determined by analysis or in many instances by reference to other sources that have either proposed or developed systems of the type herein discussed.

Spin Stabilization

Description.- Spin stabilization has been used primarily to maintain a proper direction of the rocket thrust during injection of the satellite in orbit. The spin stabilization that has incidentally resulted for the Explorer and Vanguard satellites is representative of an "open loop" type of control system. For these vehicles, the satellite spin axis initially lies close to the plane of the orbit and its orientation, in the absence of perturbations, is fixed with respect to an inertial reference system of axes. Departure from the initial heading can occur as a result of moment disturbances. The magnitude of such departure is indirectly controlled by a prior selection of the spin rate, the magnitude of which depends on the known mass properties of the satellite, and the estimated perturbations that will probably be experienced by the satellite during the mission. In order to improve this simple system, a fluid damping device can be used in the manner explained in the section entitled "Controls." Additional control elements may be further added to maintain the spin axis in some form of "closed loop" operation. This may be a controlled tracking or reorientation maneuver of some kind, utilizing the gyroscopic precessional reaction to a moment applied with respect to an inertial reference axes system. The discussion that follows will be generally applicable to an open-loop system.

L
4
3
1

The response of a spinning body to a moment disturbance that is also spinning with the body is in the form of an epicycloid. The magnitude of the epicycloid is independent of the duration of the applied moment. The number of nutations or "petals" per cycle of the epicycloid is a direct function of the satellite inertia distribution and is independent of the spin rate. The line of nodes is a circle that is the locus of the centers of all circles that describe the motion after the perturbing moment is removed. The magnitude of this final motion is a critical function of where the moment terminates on the epicycloid. The maximum angular displacement is directly proportional to the applied moment, inversely proportional to the inertia about the spin axis, and inversely proportional to the square of the spin rate (appendix A). The generalized curve of figure 3 permits a selection of design parameters for the previously described case. A similar curve can be developed for the case of an initial pitching rate.

The nutations can excite structural vibrations harmful to instrumentation and provide a mechanism for absorption of kinetic energy. This action will be destabilizing for spin about the axis of least inertia. References 11 and 12 treat additional aspects of the problem.

When the moment perturbation does not rotate with the body, as might occur from external environmental inputs, the resulting motion is, of course, precession at right angles to the applied moment. The magnitude is directly proportional to the moment and the duration and inversely proportional to the angular momentum.

Spin damping can result from interaction of an electrically conducting body shell with the magnetic field of the earth. This has been considered in references 13 and 14. The effect of spin damping is to weaken the spin stabilization until finally the satellite may tumble under the action of perturbations. The time to damp to one-half the initial spin rate may be of the order of a week to several months, depending on the satellite shape, shell material, inertia, and orbit. The strength of the earth's magnetic field at satellite altitudes is not accurately known, but may be estimated from reference 15.

L431

The earth's magnetic field can also cause a slow precession of the spinning satellite in the direction of the resultant field. A rough calculation indicates that the time to precess, say 10°, can be of the order of several months (appendix A). These effects can be offset by proper design of the satellite to reduce the magnitude of induced eddy currents in the vehicle shell or by selection of the orbit. For an extremely long mission duration any spin lost can be restored by application of a spin moment.

If it were desired to despin the satellite quickly, the method proposed by the Jet Propulsion Laboratory could be used. The method requires no prior knowledge of the spin rate and can reduce the spin rate to closely a zero value. Small weights are payed out at the end of cables that unwind around the circumference of the satellite shell under the action of centrifugal force. When the cables are radial from the center of rotation, the weights are released and the rotation of the satellite has been canceled (appendix A).

Mission suitability.- Spin stabilization offers a simple, reliable, lightweight, low-cost means of achieving a fixed orientation of one axis of a satellite vehicle. The spin axis is more easily placed initially in the plane of the orbit as illustrated in figure 4 by trajectory A. This orientation can be exploited to point the spin axis directly at some portion of the earth or to scan the earth in the manner proposed in reference 5.

A more useful orientation could be with the spin axis perpendicular to the plane of the orbit. This is illustrated by trajectory B of figure 4. This orientation could be achieved either from launch or at injection by the previously mentioned closed-loop system in which the spin axis would be precessed 90°. In this manner cameras or directional bolometers located around the circumference of the vehicle could be synchronized by horizon scanner units to the spin rate, as shown in figure 5. When pointed correctly, measurements could be recorded for use in meteorological studies and so forth. This method could provide complete global coverage by using an exactly polar orbit. An equatorial orbit could also be used, since for these two cases the relation

of the spin axis to the plane of orbit should remain unchanged. Accuracy of pointing would be difficult to evaluate but could probably be known within ±1° when the spin axis is normal to the orbital plane.

Gravitational-Gradient Method

Description.- The principle of the gravitational-gradient method has been treated in the current literature (refs. 16 and 17, for example.) The striking feature is that the method provides inherent static stabil-
ity once the satellite is properly injected into orbit. Consequently, in L
the absence of any perturbations, orientation toward the earth, as shown 4
in figure 6, can be maintained indefinitely with no expenditure of energy. 3
Figure 7 indicates two launching methods. Of practical interest are the 1
tolerances on the initial conditions and orbit eccentricity to achieve the required pointing accuracy.

The pitching motion at constant altitude for a "dumbbell" satellite configuration is independent of the two equal masses and distance between them (appendix B). For the condition of an error in pitching rate at injection, the amplitude of oscillation varies approximately as $R^{3/2}$. If this error is equal to the circular orbital rate at the particular altitude, then the amplitude is approximately 35° for all altitudes. The altitude to which this stabilization can be used depends chiefly on the stabilizing moment that can be developed to overcome estimated disturbances. An erectable structure technique could be used to extend the use of the method either to small satellites (of the order of 200 pounds) or to higher altitudes by enabling larger righting moments to be developed.

Assuming that an injection accuracy of 0.2 to 0.3 of the orbital rate is obtainable for a 500-mile circular orbit with an improved launching system, the maximum variation of orientation toward earth could be held to 6° or 10° by the launching method of figure 7(b). This might be adequate for certain missions. For the launching method of figure 7(a) the pointing accuracy would be about 35° in pitch. Additional factors such as orbit eccentricity and (for low altitudes) aerodynamic moments affecting trim attitude should be considered. Figure 8 illustrates the effect of a small orbit eccentricity for an overspeed launching condition.

It is apparent that in order to achieve a fine degree of orientation control, some form of artificial damping is needed. Damping would also permit a relaxation of launching accuracy to about 0.001 radian per second. A value of about 0.002 radian per second would cause the vehicle to cartwheel and then possibly damp to the correct or the 180° incorrect attitude. An auxiliary control system would be needed for a 180° rotation maneuver when pointed wrong. Such a game of "Sputnik Roulette"

should be avoided. The erectable structure technique can permit an additional relaxation of launching accuracy if a fast erection can be obtained and advantage taken of the inertia increase to reduce the angular rate of rotation. This appears difficult.

An active damping system would make use of a rate gyro to control a servo and inertia wheel or jet reaction torques. The signal for pitch damping must be biased to permit steady rotation in pitch to match the orbital rate ω. This implies a known circular orbit. Any error between actual and predicted orbital rate would accumulate to upset the pitch damping stabilization unless additional means are introduced to offset this tendency. Orbit eccentricity perturbations may still require close attention since their frequency is near that of the satellite's oscillations. Damping to zero amplitude may then not occur. A passive damping system may also be feasible. This is discussed further in the section entitled "Controls."

<u>Mission suitability</u>.- The gravitational method may be used with or without damping. Without damping, orientation can be held within $\pm 10^{\circ}$ for an interminable duration in the absence of disturbances provided that launching accuracy is very good and a circular orbit is achieved. Internally moving parts must be balanced. A meteorological mission or a sky survey mission can be accommodated. If an active damping system is added, the life of this damping system may be of the order of three months. With damping, a pointing accuracy of $\pm 1^{\circ}$ or better may be assured.

The method is more suitable to large elongated satellites but can be adapted to small ones using a lightweight erectable structure. The method is unsuited to a manned vehicle, unless the vehicle is extremely large.

Earth-Scanning Methods

<u>Description</u>.- Control systems that employ radiation-sensing devices for provision of earth attitude reference are attractive. Controls can be designed to handle any magnitude of perturbation anticipated at injection or later. There are at least three principal methods for scanning the earth. These may be designated as horizon scanning, area scanning, and center scanning. In the first method the effective horizon is located and compared with vehicle attitude. In the area system the projected area of the earth atmosphere is viewed and the vehicle alined with respect to the centroid of this area. In the center system radiation intensity across the earth is sensed for a maximum near the earth's center (by looking through an atmospheric "window," for example). The horizon method has received most consideration in the literature and can be accomplished in a variety of ways which differ in the type of radiation sensor, number of sensors, and scanning mode employed.

In particular, the use of a thermistor bolometer to discriminate the earth-space infrared radiation discontinuity appears feasible for continuous day and night operation. Figure 9 illustrates the basic scheme. The field of view of the detector is restricted to a narrow beam width. As the beam sweeps in attitude, the intensity of radiation contained in the beam changes sharply at the location of the effective horizon of the earth-atmosphere radiating spheroid. Figure 10 shows this effect. The curve is ideal of course. The intensity of infrared radiation arriving at the sensor depends upon the fourth power of the effective black-body temperature of the earth atmosphere ($\approx 230^\circ$ K) or space ($\approx 4^\circ$ K) sources, the proportion of these sources in the field of view of the sensor, the design of the collimating optics for the sensor (second power of the beam width and objective diameter) and the extent of solar radiation interference. The latter effect can be minimized by filtering out radiation below several microns.

L
4
3
1

If the alinement of the beam (angle A/2 in fig. 9) were to be fixed, it would correspond to a single design altitude for the satellite orbit. Variations from the design altitude would introduce "blind" zones of angular error. The angular magnitude of these zones would decrease with increase of orbit altitude if the eccentricity of the orbits remained constant. Variation of angle A/2 necessary at various altitudes has been plotted in figure 11. At an altitude of 400 miles, the angle A/2 is 65°. A deviation of ± 44 miles (which corresponds to an orbit eccentricity of $e = 0.005$) would introduce an attitude limitation of about $\pm 2^\circ$.

In order to cope with the problem of altitude variation, to permit a proportional signal for fine control and minimum power consumption, and to increase the probability of target acquisition, one of several possible scanning modes could be employed. The particular mode will vary with the number of sensors used. If a single sensor is used, the associated electronics circuits needed to process the signal are relatively complex and the stability of the electrical circuit becomes an important consideration. The sensor and its associated electronics circuits must have a low noise level to permit a favorable signal-to-noise ratio. This ratio is a factor of some uncertainty and requires determination by operation under actual or simulated space environmental conditions. The response time for the sensing element is a limiting factor in the maximum permissible scanning rate. The scanning rate must be sufficiently rapid to provide horizon-attitude signal that is effectively continuous in comparison to the frequency of the vehicle oscillations.

Control in yaw can be obtained by matching the orbital angular rate to a selected zero-yaw attitude of the vehicle by means of suitable rate-measuring devices.

The pointing accuracy of the horizon-scan method using the thermistor bolometer may be set by the troposphere. This is the effective horizon radiation source in the far infrared region. Estimates based on the effective height of the troposphere indicate a possible limiting accuracy of the order of $\pm 0.1^{\circ}$ at a satellite altitude of 500 miles. The accuracy improves for higher altitudes. A large optical system would be required to achieve this accuracy (an objective diameter of about 20 inches). It is estimated that an accuracy of $\pm 1^{\circ}$ should be obtained with a more reasonable objective diameter of 1 inch. If better accuracy is required, a possible alternative is to use a photoelectric device to supplement the thermistor bolometer during time of full daylight illumination of the earth. The error introduced by oblateness of the earth has been analyzed by reference 18 and found to be small.

L
4
3
1

Use of the cosmic-radiation-shielding effect of the earth has been proposed by Stuhlinger (ref. 19). A satellite, shielded by the earth from an assumed isotropic distribution of cosmic radiation, would detect this shielding so that a control system could be activated by discrimination of the radiation gradient. Figure 12 shows this general idea portrayed. Only the harder components of cosmic radiation would be detected by the sensors, since the softer radiation would be deflected by the earth's magnetic field. Thus, only the more energetic components are counted in coincidence. There exists a possibility that cosmic radiation could be of a directional nature rather than isotropic, because of the flatness of our galaxy, location of the earth in one of the outer spiral arms, and local radiation from the sun. This has been emphasized recently by data received from Explorer IV reported in reference 20. The intensity of cosmic radiation has been reported to vary by a factor of two depending upon orientation of the counter. The directional accuracy of the cosmic sensors, a telescope formed by several counters operating in coincidence, may be a larger factor in limiting overall control accuracy. This system should permit a rather coarse degree of attitude control.

A scheme called area scanning has been suggested and is described in the next section. Figure 13 shows three mirrors arranged in the shape of a tetrahedron with the mirrored surfaces facing the interior of sections I, II, and III. A single telescope is centrally alined to the tetrahedron. The center line of the telescope makes three equal angles with each of the planes of the mirrors. The angle between the center line of the telescope and the mirrors is such that, at the anticipated operating altitude of the satellite, each mirror, if oriented properly, can reflect a portion of the earth's surface (view A-A, fig. 13). A shutter device permits each section - I, II, or III - to be observed individually in time. The integrated response time from a mosaic of those sensors receiving infrared radiation is proportional to the area viewed rather than the intensity. By comparing the areas viewed, area differences are a measure of the verticality of the center line of the

telescope and the satellite. The accuracy will be influenced greatly by the number of sensitive elements in the mosaic, and the degree to which the time response characteristics of all the elements are identical. An additional refinement might be to vary the field of view of the telescope.

These scanning systems do not require a pitch maneuver at the end of boost for orientation toward the earth, although such a maneuver could be employed if the nose of the satellite were so designed that it had to be pointed toward the earth. The last rocket boost stage could also be spin stabilized provided the spin is later removed at final injection of the satellite into orbit. Target acquisition can be delayed until initial transients have been damped to a low level.

In addition to attitude stabilization, the sensing systems described could be further developed to provide satellite guidance for orbit control, and also guidance for the last stage of the launching vehicle near perigee. A weight saving over inertial systems could be realized.

<u>Mission suitability</u>.- Earth-scanning systems based on earth's infrared emission appear suitable for satellite missions requiring coarse to fine orientation toward the earth (to $\pm 1/2^{\circ}$). Orbits of low eccentricity ($e \leqq 0.005$) are desirable but not essential. Circular orbits of very large radii of the order of 50,000 miles can be accommodated. Beyond this distance it becomes difficult to acquire the earth in this manner. The earth must then be viewed as a point source of radiation.

Because of altitude and other probable perturbations, the system is required to operate continuously throughout the lifetime of the mission. Accuracy of control can be expected to gradually worsen as system components wear out, and the "eyes" of the system may sustain some damage by meteorites. The effective life of the control system, therefore, depends on the maximum attitude tolerance permitted by the mission. In general, a minimum lifetime of the order of three months is anticipated if enough power is available.

Moon-, Planet-, Sun-, and Star-Tracking Systems

<u>Description</u>.- The systems described in the previous section for scanning the earth can be considered to be "country cousins" to a star-tracking system in that the large reference disk of the earth is replaced by a point source of radiation, and the star-tracking system must control within much narrower limits of angular error. The design of the sensors will again depend on the temperature, size, and distance of the star or planet selected. In reference 21, for example, a beam width of 3 minutes of arc was used to scan the full moon in the infrared region. The result is shown in figure 14.

If such a body is in continuous view of the satellite, complexity in the attitude-control system is reduced. Fortunately, in the case of the sun and stars and to a limited extent the moon and planets of the solar system, this is possible. To view the sun continuously, use can be made of the nodal regression rates of selected orbits that have a yearly period. Figure 15 shows the family of circular orbits that permits continuous observation of the sun. (Also, see appendix C.) The 27.3-day sidereal period of the moon cannot be matched since the fastest nodal regression that a satellite orbit can have corresponds to twice the sidereal period and occurs for low inclination to the equator. The earth will get between the satellite and the moon in about a week (for a polar orbit). The planes of both polar ($i = 90^{\circ}$ exactly) and equatorial orbits ($i = 0^{\circ}$ exactly) remain essentially fixed in space with respect to their angular orientation. Such orbits will permit continuous sighting on a star. For example, one may select an equatorial orbit to view Polaris, the present North Star, or pick a polar orbit for Regulus, a star lying near the plane of the ecliptic. In each case the plane of these satellite orbits should preferably face the particular star. In viewing the planets, optimum conditions occur during time of closest approach to the earth. At such time the planets too can be under continuous observations.

A maneuver that initially orients the satellite is required. This maneuver can be accomplished either on the launcher or at the time the satellite is put into orbit as an additional task for an inertial guidance system located in the last booster stage of the launching vehicle. The initial heading given the satellite by this second method must be sufficiently accurate to permit the attitude-control system in the satellite to acquire and track the remote target without confusion from other bodies or stars in the apparent vicinity. The relation of the satellite orientation to both its orbit and the body selected for observation must be predicted for the point of injection into the orbit. The satellite stabilization scheme for pointing to the moon, planets, sun, or stars can remain basically the same for these distant bodies. Two such schemes are now described.

A schematic diagram of one system is shown in figure 16. In addition to the pitch and yaw stabilization indicated by the block diagram, roll-rate stabilization may also be needed. If exactly zero roll rate is specified by the mission, then it becomes necessary to simultaneously track a second body, a star. Considering only the pitch stabilization as an example, two sensors can be arranged to angles of $\pm\theta_1$ to produce the net error signal shown in figure 16(b). The magnitude of θ_1 together with the servo-gain fixes the sensitivity. One or more additional pairs of sensors can be similarly arranged to give both a broad capture capability and subsequently a much finer capability of pointing. Stabilization of this type has been developed and used very successfully

in tests at the University of Colorado and in reference 22 for taking pictures of the sun from a high altitude balloon. In the system used in these tests, values of θ_1 and θ_2 were 10° and 42.5°, respectively, and the accuracy quoted is 5 seconds to 30 seconds, depending on the servo gain.

A somewhat different scheme has been developed by reference 23 for guidance of a solar furnace. Briefly, this system uses a single detector in conjunction with a rotating shutter to generate an alternating-current error signal. The amplitude of the error signal is dependent upon the radial displacement of the radiation image from the optic axis. The phase of the error signal is dependent upon the angular position of the image. The error signal is used to operate two motors which direct the seeking system so that the optical axis is pointed toward the sun. Precautions must be taken to insure that the electrical phase matches the desired mode of operation. The tracking accuracy reported for this system is ±1 minute or better. The capture capability can be increased by widening the field of view of the seeker telescope, but at the expense of control accuracy. For conversion to satellite use, a roll control system may also be required.

The time of one week for continuous sighting of the moon using a near-polar orbit implies that more sophisticated stabilization schemes may be needed. The extent to which this may be true depends upon the magnitude of change in attitude of the satellite and position of the moon between successive sighting opportunities. If this change is small enough, periodic recapture can occur. Equatorial orbits may then be more desirable. As the moon is eclipsed the sensing units can be shut off (put to "sleep") to avoid earth radiation perturbations or capture and programed to "awaken" at the right moment for each orbit revolution of the satellite. An additional problem in tracking the moon is the variation of its surface temperature with change in phase. A programed variation in control-system parameters may be needed to match this periodic change in intensity of the moon's radiations. If intermittent loss of target can be tolerated also for the sun- and star-tracking cases, then a greater flexibility in the launching operation is achieved. Somewhat higher power level would be required for stabilization and the accuracy of tracking might be compromised when compared with that for continuous tracking.

In the case of the star-tracking mission, it is doubtful that such a mission would be seriously considered if only one star could be tracked or studied. As a minimum requirement, probably a survey capability of a local region of the celestial sphere might suffice. A possible approach might then be to select one of the brighter stars in the region as a tracking reference. The pointing of telescopic instruments could then be independently controlled by programing in accordance with precomputed offsets. If a ground-command control link were

established, the program could be more flexible. Further developement of the control system would reorient the satellite itself, in a more elaborate program which finally could lead to simultaneous viewing at a ground station.

Mission suitability.- The simple stabilization systems described in the previous section appear to be adequate for tracking a celestial body, and capable of serving as a basis for development of a more advanced control system. The system of reference 22 and of the University of Colorado is particularly attractive. The proportional-type signal should permit accurate tracking at low power level. Missions that require a large eccentricity of the orbit can be accommodated; such missions might be measuring the magnetic field and cosmic radiation out to very great distance from the earth and the relativistic effect on elliptical orbits.

Other Systems

Other systems or combinations of systems are of course possible. A hybrid system may have certain advantages over the simpler types previously considered. Several objectives in resorting to a more complex system might be the following:

(1) To maintain orientation and stability in event of temporary loss of one attitude reference

(2) To increase flexibility and broaden scope of mission, permitting change of reference bodies or simultaneous observations on more than a single reference body

(3) To improve performance

Inertial systems employing stabilized platforms or body-fixed instruments feeding a computer are limited in the satellite application by accumulation of error over a long time period and by greater weight for this type system. In larger size vehicles the weight disadvantage may be less serious. While development of exotic inertial devices having unbelievable accuracy may eventually be achieved under the condition of weightlessness, it is currently necessary to monitor and periodically correct any inertial stabilization system proposed for use in a satellite. This can be accomplished through combination with the systems already described. One possible combination would employ an automatic star tracker. Such tracking units are available and could be adapated to satellite operation, particularly for the astronomical mission of high resolution star sighting.

An interesting possibility is the combination of the earth-scan system and the gravitational-gradient system. The purpose would be to relax the accuracy requirement on the launching guidance system needed by the gravitational-gradient system alone and to obtain the indefinitely long operational lifetime capability of the gravitational system. Whether this is worthwhile would depend on the satellite size, the demand for an interminable mission, and the availability of very accurate launching vehicles. For satellites of large inertia and stabilized by a scan method, account must be taken of gravitational moments which may be stabilizing or destabilizing, depending on the design orientation of the axis of greatest inertia. This is also true, in general, for whatever other stabilization may be employed.

L
4
3
1

For altitudes up to about 300 miles it should be possible to determine vehicle attitude with respect to flight-path direction by means of atmospheric-sensitive devices. Several ionization gages might be employed with a suitable scanning action. One of the more sensitive types is the inverted magnetron gage with sensitivity to the order of 10^{-12} millimeter of mercury (ref. 24). Outgassing from the satellite may be a problem. An alternative device may be that reported in reference 25 in which molecular impact pressure (ρV^2) is measured by a microphone. A combination of apertures and detectors could be used to measure the molecular-beam direction of greatest impact pressure.

Controls

Selection.- The previous discussion has generally omitted consideration of how the attitude control torques might be developed. Several types of control (jet, flywheel, earth's magnetic field) are considered for this task. A proper selection must be made since the control unit contributes very heavily to the overall reliability, performance, weight, and power requirements of the stabilization system. Several factors that lead to the design of a specific control are: the inertia of the satellite or components to be stabilized, the initial transient maneuver to acquire the target or desired orientation, the continuous tracking maneuver during time in orbit for accomplishment of the mission, external or internal satellite perturbations (including temporary loss of reference due to an intervening body), type of internal power source available, pointing accuracy, and duration of mission.

Reference 26 presents an idealized weight comparison of attitude controls based on a severe pointing maneuver. The maneuver consisted of tracking with zero error a fixed location on the earth's surface during the time it was visible to the satellite with the attitude of the satellite initially oriented in the right direction. The assumption

of zero tracking error and zero running power for the flywheel control unit could prejudice the comparison for actual systems. The weight of a jet-reaction control, using hydrogen peroxide, was found to be either greater or less than that of a flywheel control, depending on the size of the flywheel, the electrical power source, and the number of cycles of the maneuver. Energy stored either in the form of batteries or jet propellant for the flywheel or jet control, respectively, renders that system unfit for long-term use when compared with a flywheel powered by solar batteries, for example. The power to operate the flywheel can be reduced several orders of magnitude by increasing the ratio of the flywheel inertia to satellite inertia by several orders of magnitude. The weight penalty of a large flywheel can be avoided by increasing its radius of gyration or by adding part of the satellite vehicle or equipment to the flywheel, at the same time designing to avoid aerodynamic friction losses. Saturation of the flywheel can be avoided by taking advantage of the earth magnetic damping affect or by use of intermittent jet torque pulses. Likewise, the rate of energy dissipiation of the jet propellant can be reduced by design of extendible torque nozzles at some expense of increased structure weight and reduced reliability or by selection of a propellant having higher impulse. The plasma-type motors described by reference 27 should not be ruled out for future application to satellite control mechanisms. The cross-over point for control selection must be determined for each satellite and its associated mission.

The initial transient maneuver is considerably more severe than the continuous tracking maneuver. This is shown by appendix D. A short-time, high-power control combined with a long-time, low-power control may be a desirable solution if reliability of the more complex system can be assured. Jet-type controls are desirable to remove any initial angular-momentum vector that otherwise might have to be continuously precessed. Such a control technique has been described in reference 28. In the method proposed by the reference, use is made of a jet control to handle initially the short-time, high-power requirement and a flywheel control is used for the long-time, low-power demand. Notable is the application of a magnetic drive to avoid electrical contact friction torques.

Passive damping-control methods should be mentioned, namely, fluid and spring-mass types. The application would be in damping undesirable precessional or oscillatory motions that can occur for satellites stabilized by spin or other methods, respectively. The fluid damping control has been used extensively in early ship designs. Perhaps inadvertently the viscosity and mass of bilge water was found to have a stabilizing affect on the rolling response of a ship to wave perturbations. Reference 29 lists some variations in the application to ships. It is recommended that mercury be used for damping the motions of satellite vehicles. The great density and viscosity of mercury should permit its effective

use in a fluid damper control. The weight of mercury must be minimized by a testing program. In the case of those satellites that may use mercury (or some other fluid) in a steam cycle for internal power development, additional weight of mercury may not be required for a passive damping control. The spring-mass damping control operates by resonant excitation of the small control mass or masses employed for the purpose. This method may be more effective (and sophisticated) than the fluid-alone type and more amenable to analytical analysis. Passive damping controls should be very reliable over long periods of time and require no power in contrast to active types. When used in spin stabilization, the spin axis must be the axis of greatest inertia; otherwise, the absorption of kinetic energy by the damper will be destabilizing. The mode may change to spin about the axis of greatest inertia, although the angular momentum vector will be unchanged.

<u>Reliability</u>.- The greatest uncertainty in the selection of a control type is the factor of reliability as affected by the initial launching environment and subsequent long-term exposure to the space environment. Among the many questions that might be asked are the following. How long can chemicals be stored for use in jet controls and how long can regulation of fluid flow to the jet nozzle be maintained? How much will the condition of weightlessness extend the service life of control components and change the control performance from that determined at 1 g? What is the tolerance of the control types to launching accelerations and vibrations and to temperature? Partial answers to some of the questions concerning reliability can be determined by ground testing of control components, control units, and, in fact, the complete stabilization system. The latter test requires development of a near-frictionless support for the test vehicle and simulated targets for determination of reliability and performance at 1 g. This can be done. However, it is obvious that a system tested for a long period of time may be more susceptible to failure and the reliability of a fresh system selected for flight is not completely assured by the more severe testing of its supposedly identical twin. The final proof is still in the orbiting.

STABILIZATION-SYSTEM COMPARISONS

The several stabilization systems are compared in simplest form with regard to weight and power, development effort and reliability, and control accuracy. The initial transient maneuver was selected as a basis for weight and power estimates of gravitational and radiation-sensing systems using a flywheel-control unit. (See appendix E.) Admittedly the estimates do not represent optimum systems but may serve to establish trends on an order of magnitude basis.

Weight and Power

Table VI lists the several sizes of satellites considered. For the vehicles stabilized by gravitational gradient, use is made of the last stage booster to augment the static restoring moment. For satellites weighing less than 3,000 pounds, a lightweight erectable structre was assumed to permit a maximum static-moment capability of 1 inch-ounce for all sizes. Whether this magnitude of static moment is required of the smaller vehicles would depend on a more refined analysis of possible disturbances for the particular payload. For vehicles stabilized by radiation-sensing systems, the vehicle inertia is desirably small for the capture or steady tracking maneuvers required of the control system. For these vehicles the last-stage booster unit is assumed to be separated at injection of the satellite. For spin-stabilized vehicles most of the spin can be accomplished on the ground, or during the first-stage boost. The spin-up power and weight associated with this power that should be charged to the orbiting satellite can therefore be small. A weight penalty may be incurred in mounting the vehicle to permit spin-up. For the present comparisons no weight or power penalty is imposed on spin stabilization. Likewise the weight required for power supply is not included for any case. In regard to this last item, it would be possible to divert enough of the payload instrumentation power to the stabilization system when the latter is in trouble as may occur during the initial transient maneuver or as a result of an infrequent perturbation of large magnitude. At such times measurements recorded by some of the instrumentation would probably be of little value anyhow.

Tables VII and VIII present design conditions for some possible initial launch and transient conditions. A factor of two was used in conjunction with the launching-rate error to increase the probability of a successful capture by the satellite stabilization system. This fair launch with an error of 0.0010 radian/sec was used for further design estimates. It is noted that an error of this magnitude would work the gravitational system near its maximum capability (before tumbling). For earth-scan systems an angle of 63.6° was used (linear system) since this is comparable to the 85° capability of the gravitational system (nonlinear). For a sun-tracking system it was assumed that this large initial angle was also applicable, but for a star-tracking system the initial deviation was taken to be 10°.

Table IX presents weight and power for some components of a stabilization system which are independent of the size of the satellite. Floated rate gyros were chosen as being currently available, capable of the accuracy required for rate measurement, and efficiently packaged. The smallest gyro of this type weighs about 0.5 pound but is somewhat less accurate, while the largest is about 3 pounds. Air-bearing gyros

are not available in comparable weights and would require a continuous supply of gas for the bearing. The disadvantage of the gyros is the power required to maintain running speed and for the floated gyro, the power required to maintain the 165° F operating temperature. Recent improvement in gyro design may avoid the last item. Fortunately for sun-, star-, and some versions of earth-scanning systems, rate gyros are not required to furnish the rate signal; however, if the satellite is to be recovered, miniature rate gyros may be needed to damp oscillations on reentry.

Estimated weight and power for the active damping system of a satellite stabilized by gravitational gradient are tabulated in table X. It is possible, for example, to eliminate some components by cross-coupling the roll-yaw axes provided that the satellite has pitch, roll, and yaw static stability. One of the greatest uncertainties in the estimate is the flywheel and drive unit, and in particular the power required to sustain the maximum running speed. For the present estimate, rough estimates were made on the basis of running power requirements for floated rate gyros. Table XI lists the weight and power requirements for earth-scanning systems with and without rate gyros. The smaller satellites require less weight and power by virtue of the smaller control unit required.

A summary of weight and power requirements for the various systems is presented in tables XII and XIII and by figures 17 and 18. Spin and gravity stabilization without damping require minimum weight and power. For satellites of the order of 3,000 pounds, the gravitational-gradient method with active damping requires less weight and power than radiation-sensing types of stabilization.

Development Effort and Reliability

The relative effort required to develop the various stabilization systems is noted in table XIV. Spin stabilization is believed to require the least effort. For a large satellite stabilized by the gravitational method and no damping, the effort should also be a minimum. However, in this case, effort may be required on the launching vehicle to improve accuracy of satellite injection.

The relative reliability estimate is presented in table XV. The estimate assumes the satellite to be properly placed in orbit. On this basis the system that is most reliable is the gravity system with no damping. Natural examples of the possible magnitude of reliability for stabilization in this manner are the moon and the planet Mercury which constantly present the same face toward the parent body. With an active damping system the reliability or useful life of the damping system could possibly be extended by the expedient of shutting off the damping control unit during periods of accurate tracking with small deviations.

Control Accuracy

The accuracy for systems having no damping is largely a function of injection accuracy of the launching vehicle. The system that suffers most in this regard is the gravity system. The spin and radiation-sensing methods are sufficiently flexible that allowance can be made during design of the system or changes made later in the program. With damping added, the gravity method is still hypersensitive to the magnitude of the launching-rate error that may cause tumbling. Presumably with damping, initial oscillations can be reduced to the order of 1° or less for all systems. The minimum value will depend on limit-cycle performance of the stabilization system. This is a matter for detailed servomechanism-system analysis. In general, a system employing an on-off type of control signal or control torque would be expected to have larger limit-cycle oscillations than a system employing a proportional signal. The condition of weightlessness may be expected to improve performance by reducing "stickiness" of control elements. Performance determined by ground tests at 1 g may be conservative. Systems for pointing to earth are subject to disturbances arising from a close earth-to-satellite relationship that tend to reduce accuracy of pointing. Systems for pointing to other celestial bodies can be considerably more accurate, but then they have to be when the mission requires sun- or star-tracking measurements. The relative pointing accuracy for different systems is listed in table XVI.

L
4
3
1

CONCLUDING REMARKS

As a result of this survey of possible attitude stabilization and control methods, the following observations were made. However, more detailed analysis and testing are needed to accurately establish the weight, power, performance, and reliability of the various systems that have been considered.

1. Most of the mission-orientation requirements can be initially met by relatively simple lightweight stabilization systems.

2. An improved version of spin stabilization incorporating earth-horizon scan units is attractive for meteorological payloads.

3. The gravitational-gradient method is attractive for large satellites if the launching-system accuracy at injection of the satellite is no worse than 0.001 radian/sec.

4. Passive damping-control units employing either mercury or resonant spring-mass systems should be developed in order to reduce power required and increase reliability.

5. Systems employing radiation sensing units can be quite versatile in meeting performance requirements with adequate useful life and moderate weight and power and are also compatible with the launching vehicle capability. Such systems should be developed.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., June 5, 1959.

L
4
3
1

APPENDIX A

STABILITY OF SPINNING BODIES

Equations of angular motion referred to an inertial axis system for a spinning body during application of a moment about a body axis are given as follows:

L 4 3 1

For small angles of θ and ψ and for $\dot{\phi} \approx p = \text{Constant}$,

$$\phi = pt \tag{1}$$

$$\theta = \frac{M}{I_X p^2\left(1 - \frac{I_X}{I_Y}\right)}\left(\frac{I_X}{I_Y} \sin pt - \sin \frac{I_X}{I_Y} pt\right) \tag{2}$$

$$\psi = \frac{M}{I_X p^2\left(1 - \frac{I_X}{I_Y}\right)}\left[\frac{I_X}{I_Y}(1 - \cos pt) - \left(1 - \cos \frac{I_X}{I_Y} pt\right)\right] \tag{3}$$

$$\theta = \left[\frac{M\sqrt{2(1 - \cos p)\left(1 - \frac{I_X}{I_Y}\right)\tau}}{I_X p^2\left(1 - \frac{I_X}{I_Y}\right)}\right]\sin\left(\frac{I_X}{I_Y} pt_t + \beta_1\right) + \frac{M}{I_X p^2} \sin p\tau \tag{4}$$

$$\psi = \left[\frac{M\sqrt{2(1 - \cos p)\left(1 - \frac{I_X}{I_Y}\right)\tau}}{I_X p^2\left(1 - \frac{I_X}{I_Y}\right)}\right]\sin\left(\frac{I_X}{I_Y} pt_t + \beta_2\right) - \frac{M}{I_X p^2}(1 - \cos p\tau) \tag{5}$$

where

$$\beta_1 = p\tau + \tan^{-1}\left(\frac{\sin \omega\tau}{1 - \cos \omega\tau}\right)$$

$$\beta_2 = p\tau + \tan^{-1}\left(\frac{1 - \cos \omega\tau}{\sin \omega\tau}\right)$$

and

$$\omega = p\left(1 - \frac{I_X}{I_Y}\right)$$

An approximation of the maximum angular displacement may be obtained by

$$\frac{2M}{I_X p^2 1 - \frac{I_X}{I_Y}} + \frac{M}{I_X p^2}\left[1 - \cos\left(\frac{\pi}{1 - \frac{I_X}{I_Y}}\right)\right] \tag{6}$$

In the previous equations the moments of inertia and the applied moment are assumed constant with the moment acting about the body yaw axis and rotating with the body. Equations (1), (2), and (3) apply during application of the moment, and the initial conditions other than p are assumed to be zero. Equations (4) and (5) apply after termination of the moment. Figure 3 presents the maximum displacement parameter. Calculated results for specific cases were in good agreement with digital computations.

For the precessional effect of a magnetic field on a spinning cylinder, assume that the body is rotating at a fixed rotational speed at an angle to the resultant magnetic field. The expression for the current i generated in the spinning body shell is:

$$di = \frac{\mu H \sin \xi l p \sigma r^2 dr \sin \zeta \, d\zeta}{l + r_o + r_i}$$

Besides a torque created tending to slow the spin down, another torque is also created from the interaction of the magnetic field of the induced current flowing in the ends of the cylinder and the magnetic-field component that is parallel to the spin axis. The expression for the end force is:

$$dF = \frac{(\mu H)^2 \sin 2\xi l p \sigma r^3 dr \sin^2 \zeta \, d\zeta}{l + r_o + r_i}$$

A couple is created which precesses the body into the resultant magnetic-field vector. The expression for the precessional motion is:

$$\dot{\xi} = \frac{\pi}{8} \frac{(\mu H)^2 \sin 2\xi l^2 \sigma \left(r_o^4 - r_i^4\right)}{I_X\left(l + r_o + r_i\right)}$$

A method for despinning a spinning vehicle after achieving orbit is as follows. A weight is attached at the end of a flexible cable of a predetermined length. Two such cables are wrapped around the circumference of a vehicle. The length of the cable required is completely independent of the spin as seen by the following:

$$L = \sqrt{\frac{I_X}{m} + r_o^2} - r_o$$

L
4
3
1

As the body is spinning in orbit the weights are extended from the circumference of the satellite by centrifugal force. At the instant the cables extend radially from the spin axis, they are released. At this time the spin is closely zero. This method can also be used to obtain a reduced rate of spin.

APPENDIX B

PITCH RESPONSE FOR SATELLITE STABILIZED BY GRAVITATIONAL-GRADIENT METHOD, NO DAMPING

In pitch the expression for the gravitational moment is:

$$T = 3 \frac{\mu}{R^3} \sum_{n=1}^{i} m_n d_n^2 \sin 2\alpha_n$$

If the satellite has the shape and mass distribution of a sphere, T is equal to 0. For the simple case of two concentrated masses m located a distance of 2d apart,

$$T = 3 \frac{\mu}{R^3} md^2 \sin 2\alpha$$

where $\mu = 1.41 \times 10^{16}$ ft^3/sec^2 and R is the distance of the satellite from the center of the earth with respect to inertial axes

$$T = I\ddot{\theta}$$

For the case under consideration,

$$I = 2md^2$$

Hence for a "dumbbell," the equation reduces to

$$\frac{3\mu}{R^3} \sin 2\alpha = 2\ddot{\theta}$$

and the motion is independent of the distance d and mass m of the satellite; however, the moment developed at angle α is not. The relation between θ and α for circular orbits is $\omega t = \theta + \alpha$ where $\omega^2 = \frac{\mu}{R^3}$. If the assumption $\sin 2\alpha = 2\alpha$ is used, the equation reduces further to

$$\ddot{\alpha} + \frac{3\mu}{R^3} \alpha = 0$$

L
4
3
1

since

$$\ddot{\theta} = -\ddot{\alpha}$$

The solution is

$$\alpha = \alpha_o \cos t\sqrt{\frac{3\mu}{R^3}} + \dot{\alpha}_o\sqrt{\frac{R^3}{3\mu}} \sin t\sqrt{\frac{3\mu}{R^3}}$$

L
4
3
1

For results presented, the linearization was not made except where noted. Solutions were obtained with digital computer.

The following cases were analyzed for circular orbits to investigate orbital injection of a satellite stabilized in this manner:

Effect of altitude with initial conditions of $\alpha = \theta = \dot{\theta} = 0$ and $\dot{\alpha} = \omega$

Effect of initial θ (and α) for initial values of $\dot{\theta} = 0$ and $\dot{\alpha} = \omega$ at a constant altitude of 500 miles

Effect of initial $\dot{\theta}$ (and $\dot{\alpha}$) for initial values of $\alpha = \theta = 0$ at a constant altitude of 500 miles

For noncircular orbits with ellipticity e

$$R = \frac{(r + h_o)(1 + e)}{1 + e \cos \gamma}$$

and

$$\gamma = \alpha + \theta$$

$$\dot{\gamma} = \frac{\mu^{1/2}}{\left[(r + h\ )(1 + e)\right]^{3/2}}(1 + e \cos \gamma)^2 = \dot{\alpha} + \dot{\theta}$$

$$\ddot{\gamma} = \frac{-2\mu(1 + e \cos \gamma)^3(e \sin \gamma)}{(r + h_o)(1 + e)^3} = \ddot{\alpha} + \ddot{\theta}$$

where r is the radius of the earth.

The following cases were analyzed for noncircular orbits with a digital computer:

Effect of ellipticity e for initial values of $\alpha_o = \theta_o = 0$ and $\dot{\theta}_o = \omega = \sqrt{\frac{\mu}{R^3}} = \sqrt{\frac{\mu}{(r + h_o)^3}}$ at $h_o = 500$ miles

Effect of $\dot{\theta}_o$ for $\alpha_o = \theta_o = 0$ and $e = 0.005$, 0.01, 0.05, and 0.10 at $h_o = 500$ miles

The results are plotted in figures 8, 19, and 20.

L
4
3
1

APPENDIX C

CIRCULAR ORBITS THAT PERMIT A CONTINUOUS VIEW OF THE SUN

It is first necessary to calculate the family of orbits having the same regression rate as the orbital rate of the earth around the sun. This rate is 2π radians per year or 0.199×10^{-6} radian/sec. The calculation proceeds in accordance with reference 30. The expression for the regression of nodes is

$$n = x\omega \cos i$$

where

$$x = \frac{3(C - A)}{2MR^2}$$

and

$$\frac{3(C - A)}{2Mr^2} = 1.64 \times 10^{-3}$$

$$\omega \approx \sqrt{\frac{\mu}{R^3}}$$

This results in

$$\cos i = \frac{0.0979}{\left(\frac{r}{R}\right)^3}$$

The effect of the $23^{\circ}27'$ tilt of the earth's axis (N-S) to the plane of the ecliptic was determined graphically. Approximate cutoff points are noted in figure 15.

APPENDIX D

MOMENTUM, TORQUE, AND POWER OF ANGULAR MANEUVERS

Possible angular maneuvers include those of a short time, transient nature such as may occur at injection of the satellite as it separates from the launching vehicle, those of a purposely recurring or intermittent nature such as tracking a point on the earth's surface, and those of a steady or continuous nature as may be imposed by such factors as aberration of light in tracking a star with extreme precision, oblateness of the earth or orbit eccentricity in pointing toward the earth's center. It is necessary to consider the maximum specific momentum, torque, and power occasioned by these maneuvers in order to establish the weight and power requirements for a control system. The maximum specific angular moment is defined as

$$\left(\frac{\text{Angular momentum}}{I}\right)_{max} = |\dot{\theta}|_{max}$$

the maximum specific torque is

$$\left(\frac{\text{Torque}}{I}\right)_{max} = |\ddot{\theta}|_{max}$$

and the maximum specific power is

$$\left(\frac{\text{Power}}{I}\right)_{max} = 1.356\,|\dot{\theta}\ddot{\theta}|_{max}$$

For the present purpose rather crude estimates will suffice to indicate relative levels.

Transient Maneuver

Two aspects of the transient maneuver are of interest. In the first instance only the values associated with damping the motion are to be determined. In the second instance the maximum values for the maneuver are required when both static and dynamic stability are supplied by the stabilization system. The initial conditions or errors to be reduced by the satellite stabilization are θ_o and $\dot{\theta}_o$. A first assumption is that θ_o is negligible and that $\dot{\theta}_o$ is indeed closely equal to $\dot{\theta}_{max}$. The maximum angular momentum, torque, and power occur during the first

oscillation. Assume that

$$\theta = \frac{\dot{\theta}_o}{a} \sin at$$

where

$$a = \frac{2\pi}{\text{Period}}$$

from which the first θ_{max} can be determined and that subsequently

$$\theta = \theta_{max} e^{-bt} \cos at$$

where

$$b = \frac{2.303}{t_{1/10}}$$

With these assumptions, the maximum specific angular momentum is

$$\left(\frac{\text{Angular momentum}}{I}\right)_{max} = \left|\dot{\theta}_o\right|$$

The maximum specific torque is

$$\left(\frac{\text{Torque}}{I}\right)_{max} = \left|\frac{2\pi\dot{\theta}_o}{\text{Period}}\right|$$

and the maximum specific power is

$$\left(\frac{\text{Power}}{I}\right)_{max} = \frac{1.356\pi}{\text{Period}} \dot{\theta}_o^{\,2}$$

For the damping portion of the maneuver the maximum specific angular momentum eventually absorbed is also

$$\left(\frac{\text{Angular momentum}}{I}\right)_{max} = \left|\dot{\theta}_o\right|$$

However the maximum specific damping torque is only

$$\left(\frac{\text{Torque}}{I}\right)_{max} = \left|b^2\theta\right|_{max}$$

and the maximum specific damping power is only

$$\left(\frac{\text{Power}}{I}\right)_{max} = 1.356 b^3 \theta^2_{max}$$

Recurring or Intermittent Maneuver

In the category of recurring or intermittent maneuvers, the maneuver of reference 26 is of interest. The maximum specific angular momentum is

$$\left(\frac{\text{Angular momentum}}{I}\right)_{max} = \frac{\mu^{1/2}}{hR^{1/2}}$$

The maximum specific torque is

$$\left(\frac{\text{Torque}}{I}\right)_{max} = \left|\frac{\mu hr(r + R)\sin \gamma}{R^2(2rR + h^2 - 2rR \cos \gamma)^2}\right|_{max}$$

for

$$\cos \gamma = \frac{1 + \sin^2\gamma}{1 + \frac{h^2}{2rR}}$$

and the maximum specific power is

$$\left(\frac{\text{Power}}{I}\right)_{max} = 1.356\left|\frac{-\mu^{3/2}hrR^2(r + R)(R - r \cos \gamma)\sin \gamma}{(2rR + h^2 - 2rR \cos \gamma)^3}\right|_{max}$$

for

$$\cos \gamma = \frac{(6rR^2 - 2r^2R - h^2r) + (4r^2R + 2h^2r - 4rR^2)\cos^2\gamma + 2r^2R \cos^3\gamma}{2rR^2 + h^2R + 4r^2R}$$

Continuous Tracking Maneuver

Consider the possibility of an overspeed eccentric orbit. The design value of initial pitching rate for the satellite is ω. The

error in matching the actual variable orbital rate is

$$\dot{\theta} = \frac{\mu^{1/2}(1 + e \cos \gamma)^2}{\left[(r + h_o)(1 + e)\right]^{3/2}} - \omega$$

where $\dot{\gamma} = \dot{\theta} + \omega$. The specific angular momentum is $\dot{\theta}$. The value of ω should be selected to minimize either momentum or power or the best compromise thereof. In reality ω would probably be selected to match some circular design orbit. In any case the specific control torque is

$$\left(\frac{\text{Torque}}{I}\right) = \frac{-2\mu(1 + e \cos \gamma)^3(e \sin \gamma)}{\left[(r + h_o)(1 + e)\right]^3}$$

and the maximum occurs for

$$\cos \gamma = 4e \sin^2\gamma = e$$

The specific power is

$$\left(\frac{\text{Power}}{I}\right) = 1.356\left[\frac{-2\mu^{3/2}(1 + e \cos \gamma)^5(e \sin \gamma)}{\left[(r + h_o)(1 + e)\right]^{9/2}} + \frac{2\omega\mu(1 + e \cos \gamma)^3(e \sin \gamma)}{\left[(r + h_o)(1 + e)\right]^3}\right]$$

and the maximum occurs for

$$\omega(\cos \gamma - 4e \sin^2\gamma + e) = \frac{\mu^{1/2}}{\left[(r + h_o)(1 + e)\right]^{3/2}}\Big[\cos \gamma(1 + e \cos \gamma)^3 - 5e \sin^2\gamma(1 + e \cos \gamma)^2\Big]$$

Examples

Initial transient maneuver.- The following launching vehicle capabilities are assumed (ref. 31) for an initial transient maneuver:

$$\theta_o = 0.005 \text{ radian/sec} \qquad \text{(sloppy launch)}$$

$$\theta_o = 0.0005 \text{ radian/sec} \qquad \text{(good launch)}$$

to which is applied a factor of two in conjunction with initial maximum amplitudes of 10°, 31.8°, and 85°, and time to damp to one-tenth amplitude

is taken to be 6 hours. For radiation-sensing systems the period is

$$\text{Period} = \frac{2\pi\theta_{max}}{\dot{\theta}_o}$$

These conditions are summarized in the following table:

$\dot{\theta}_o$, radian/sec	Period, sec, for θ_{max} of -		
	10°	31.8°	85°
0.0010	1,095	3,480	9,310
.010	109.5	348.0	931.0

L
4
3
1

For a gravitational-gradient system operating at a circular altitude of 500 miles the period is 3,480 seconds and for a θ_{max} of 85° the maximum allowable $\dot{\theta}_o$ is about 0.002 radian/sec (0.0027 if it were linear). With $\dot{\theta}_o = 0.002$ and a period of 3,480 seconds, θ_{max} equals 63.6° when linear. For values of $\dot{\theta}_o > 0.002$ the satellite stabilized by gravity would cartwheel.

The maximum specific values for radiation-sensing systems are:

θ_{max}, deg	Momentum, 1/sec	Torque, 1/sec^2	Power, $\frac{\text{watts}}{\text{slug-ft}^2}$
$\dot{\theta}_o = 0.0010 = 2$ times probable value (good launch)			
10	0.0010	5.73×10^{-6}	3.88×10^{-9}
31.8	.0010	1.80	1.22
85	.0010	.67	.46
$\dot{\theta}_o = 0.0020$ (fair launch)			
10	0.0020	22.92×10^{-6}	31.0×10^{-9}
63.6	.0020	3.61	4.88
$\dot{\theta}_o = 0.010$ (sloppy launch)			
10	0.010	5.73×10^{-4}	3.88×10^{-6}
31.8	.010	1.80	1.22
85	.010	.67	.46

The maximum specific values associated with damping are (for $t_{1/10}$ = 6 hours):

$\dot{\theta}_o$, radian/sec	θ_{max}, deg	Momentum, 1/sec	Torque, 1/sec^2	Power, $\frac{\text{watts}}{\text{slug-ft}^2}$
0.0010	31.8	0.0010	6.35×10^{-9}	5.13×10^{-13}
.0020	63.6	.0020	12.70	20.5

L
4
3
1

Intermittent maneuver.- Maximum specific values associated with an intermittent maneuver are:

Circular altitude, miles	Angular momentum, 1/sec	Torque, 1/sec^2	Power, $\frac{\text{watts}}{\text{slug-ft}^{-2}}$
300	15.0×10^{-3}	13.6×10^{-5}	21.7×10^{-7}
500	9.4	5.1	5.1
1,000	4.5	.98	.51
10,000	.1	.001	.00005

Continuous tracking maneuver.- Maximum specific values associated with a continuous tracking maneuver for injection at an altitude of 500 miles and overspeed elliptical orbits with ω = 0.00104 radian/sec are:

Eccentricity, e	Angular momentum, 1/sec	Torque, 1/sec^2	Power, $\frac{\text{watts}}{\text{slug-ft}^2}$
0.01	2×10^{-5}	2×10^{-8}	1×10^{-11}
.05	11	11	4
.10	22	22	8

APPENDIX E

WEIGHT AND POWER REQUIRED BY FLYWHEEL CONTROL UNITS

The initial transient maneuver at satellite injection is used as a basis for estimates. The control weight and power depend on the maneuver maximum angular momentum, torque, and power. In addition, power is required to maintain the maximum estimated running speed of the flywheel and drive motor combination. This latter item can be estimated roughly from data available on rate gyroscopes operating at 1 g. For zero g the power may be less, depending on relative contributions from friction, windage, and eddy currents in the drive rotor. Friction can be reduced by minimizing the bearing preload. Windage can be reduced by operating the flywheel at a minimum of static pressure surrounding the flywheel (by exposing the flywheel to the vacuum of space). A damping flywheel must eventually absorb the initial angular momentum error of the satellite. Thus

$$\left(I_R\dot{\theta}_R\right)_{final} = \left(I_S\dot{\theta}_S\right)_o$$

when

$$\left(\dot{\theta}_{R,o}\right) = 0 = \theta_S$$

When the flywheel also provides static stability, its maximum momentum is $\approx 2\left(I_S\dot{\theta}_S\right)_o$ at $\dot{\theta}_S \approx -\dot{\theta}_{S,o}$ and $I_R\dot{\theta}_R = \left(I_S\dot{\theta}_S\right)_o$ at $\theta_{max,o}$. Allowance should be made for cumulative perturbation effects that may tend to saturate the flywheel to its limiting speed over the time required for the mission. This factor will not be included in the present instance.

The maximum torque required of the drive motor in the absence of other disturbances is

$$(\text{Torque})_{max} = \left|I_R\ddot{\theta}_R + \text{Running torque at } \dot{\theta}_R\right|_{max}$$

$$= \left|I_S\ddot{\theta}_S + \text{Running torque at } \dot{\theta}_R\right|_{max}$$

The maximum power required of the drive motor operating at K, percent efficiency, is

$$(\text{Power})_{max} = \left|\frac{I_S}{I_R}\,\frac{(\text{Maneuver power})}{K} + \text{Running power}\right|_{max}$$

For an active damping control, the maximum running power governs since the maneuver power is generally quite small and approaches zero as the running power approaches maximum. For a radiation-sensing system (with static stability from the flywheel) the maximum power occurs either at the initial peak amplitude and is the sum of the maximum maneuver power plus the running power or at $\dot{\theta}_S = -\dot{\theta}_{S,o}$ for zero maneuver power and maximum running power.

The design of the flywheel is influenced by such factors as limiting speed (from the standpoint of bearing wear, running power, and flywheel stress), size, and shape (ruggedness and packaging). For the present estimates, a limiting design speed of 12,000 rpm was arbitrarily selected. For angular momentum above 8 $\frac{\text{slug-ft}^2}{\text{sec}}$ the radius was either held constant at 2 inches or decreased to maintain constant stress. Ninety percent of the flywheel weight was assumed in the rim. The drive motors were assumed to be 400-cycle-per-second aircraft type, with efficiency varying from about 30 percent for the smallest to 50 percent for the largest. Some general expressions relating flywheel characteristics to angular momentum are:

For constant rotational speed and radius (12,000 rpm and 2 inches, respectively),

$$W_R = \frac{0.923}{0.9}\left(Ip_S\right)_{max}$$

For constant rotational speed (12,000 rpm),

$$W_R = Kr_R{}^3$$

(where r_R is radius of flywheel) and a radius of 2 inches at 12,000 rpm

$$W_R = \frac{2.12}{0.9}\left[\left(Ip_S\right)_{max}\right]^{3/5}$$

$$r_R = 0.110\left[\left(Ip_S\right)_{max}\right]^{1/5}$$

For constant stress with a radius of 2 inches at 12,000 rpm and an angular momentum of 8 $\frac{\text{slug-ft}^2}{\text{sec}}$,

$$W_R = Kr_R{}^3$$

$$pr_R = 209$$

$$W_R = \frac{1.545}{0.9}\left[(Ip)_{max}\right]^{3/4}$$

$$r_R = 0.099\left[(Ip)_{max}\right]^{1/4}$$

$$p = \frac{2110}{\left[(Ip)_{max}\right]^{1/4}}$$

REFERENCES

1. Haviland, R. P.: Reliability in Guided Missiles. Jet Propulsion, vol. 25, no. 7, July 1955, pp. 321-325, 330.

2. Lawrence, H. R., and Amster, W. H.: Reliability Achievement and Demonstration in a Development Program. [Preprint] 632-58, presented at Am. Rocket Soc. Semi-Annual Meeting (Los Angeles, Calif.), June 9-12, 1958.

3. Roberson, R. E.: An Approach to System Performance Prediction - I. Reliability as an Evaluation Factor. [Preprint] 656-58, presented at Am. Rocket Soc. Fall Meeting (Detroit, Mich.), Sept. 15-18, 1958.

4. ARS Space Flight Committee: On the Utility of an Artificial Unmanned Earth Satellite. Jet Propulsion, vol. 25, no. 2, Feb. 1955, pp. 71-78.

5. Van Allen, James A., ed.: Scientific Uses of Earth Satellites. Univ. of Michigan Press (Ann Arbor), c.1956.

6. Technical Panel on the Earth Satellite Program: Basic Objectives of a Continuing Program of Scientific Research in Outer Space. U.S. Nat. Committee for Int. Geophys. Year 1957-58, Nat. Acad. Sci. (Washington, D. C.), Feb. 14, 1958.

7. Anon.: NASA Details Space Contracts Program. Aviation Week, vol. 69, no. 6, Aug. 11, 1958, pp. 36-41.

8. Glaser, Arnold Henry: Meteorological Utilization of Images of the Earth's Surface Transmitted From a Satellite Vehicle. AFCRC-TR-57-241 (ASTIA Doc. No. AD 146764), Air Force Cambridge Res. Center, Oct. 31, 1957.

9. Prew, Henry E.: Data Links for Space Exploration - Their Nature, Applications and Limitations. Proc. Am. Astronautical Soc., 4th Annual Meeting (New York), Jan. 1958, pp. 26-1 - 26-11.

10. Pierce, J. R.: Orbital Radio Relays. Jet Propulsion, vol. 25, no. 4, Apr. 1955, pp. 153-157.

11. Laitone, E. V.: Spin-Stabilized Unsymmetrical Bodies. Jet Propulsion (Tech. Notes), vol. 27, no. 4, Apr. 1957, pp. 417-419.

12. Jarmolow, Kenneth: Dynamics of a Spinning Rocket With Varying Inertia and Applied Moment. Jour. Appl. Phys., vol. 28, no. 3, Mar. 1957, pp. 308-313.

13. Hooper, John W.: The Damping of Metallic Cylindrical and Spherical Bodies Rotating in a Uniform Magnetic Field. Tech. Rep. DG-R-15, Dev. Operations Div., Army Ballistic Missile Agency (Huntsville, Ala.), Nov. 1957.

14. Rosenstock, Herbert B.: The Effect of the Earth's Magnetic Field on the Spin of the Satellite. Astronautica Acta, Vol. III, Fasc. 3, 1957, pp. 215-221.

15. Fougere, P., and McClay, J.: Preparation of Mutually Consistent Magnetic Charts. Geophys. Res. Paper No. 55 (AFCRC-TR-57-225, ASTIA Doc. No. AD 117268), Air Force Cambridge Res. Center, June 1957.

16. Roberson, Robert E.: Gravitational Torque on a Satellite Vehicle. Jour. Franklin Inst., vol. 265, no. 1, Jan. 1958, pp. 13-22.

17. Davis, William R.: Determination of a Unique Attitude for an Earth Satellite. Proc. Am. Astronautical Soc. 4th Annual Meeting (New York), Jan. 1958, pp. 10-1 - 10-15.

18. Roberson, Robert E.: Optical Determination of Orientation and Position Near a Planet. [Preprint] 643-58, presented at Am. Rocket Soc. Semi-Annual Meeting (Los Angeles, Calif.), June 9-12, 1958.

19. Stuhlinger, Ernst: Control and Power Supply Problems of Instrumented Satellites. Inst. Radio Eng., Trans. on Instrumentation, PGI-5, 1956, pp. 19-26. (Also available as [Preprint] 279-55, presented at the Am. Rocket Soc. Twenty-Fifth Annual Anniversary Meeting (Chicago, Ill.), Nov. 14-18, 1955.)

20. Klass, Philip J.: Radiation Poses Manned Space Problem. Aviation Week, vol. 69, no. 7, Aug. 18, 1958, pp. 32-33.

21. Anon.: Survey of Sky Radiation Intensities in the Thermal Region. NAVORD Rep. 4869, U.S. Dept. Navy, Apr. 5, 1957.

22. Ross, Malcolm D.: Plastic Balloons for Planetary Research. Proc. Am. Astronautical Soc. 4th Annual Meeting (New York), Jan. 1958, pp. 42-1 - 42-13.

23. Moore, James G., and St. Amand, Pierre: A Guidance System for a Solar Furnace. NAVORD Rep. No. 5846 (NOTS 1943), U.S. Naval Ord. Test Station (China Lake, Calif.), Feb. 25, 1958. (Also published in Jour. Solar Energy Sci. and Eng., vol. 1, no. 4, Oct. 1957, pp. 27-29.)

24. Hobson, J. P., and Redhead, P. A.: Operation of an Inverted-Magnetron Gauge in the Pressure Range 10^{-3} to 10^{-12} mm. Hg. Canadian Jour. of Phys., vol. 36, no. 3, Mar. 1958, pp. 271-288.

25. Dessler, A. J., Hanson, W. B., et al.: A New Instrument for Measuring Atmospheric Density and Temperature at Satellite Altitudes. Proc. Am. Astronautical Soc., Western Regional Meeting (Palo Alto, Calif.), Aug. 1958, pp. 24-1 - 24-3.

26. Adams, James J., and Chilton, Robert G.: A Weight Comparison of
L Several Attitude Controls for Satellites. NASA MEMO 12-30-58L,
4 1959.
3
1 27. Bostick, Winston H.: Plasma Motors. Proc. Am. Astronautical Soc., 4th Annual Meeting (New York), Jan. 1958, pp. 2-1 - 2-11.

28. Haeussermann, Walter: An Attitude Control System for Space Vehicles. [Preprint] 642-58, presented at the Am. Rocket Soc. Semi-Annual Meeting (Los Angeles, Calif.), June 9-12, 1958.

29. Chadwick, Joseph H., Jr.: Ship Stabilization in the Large: A General Analysis of Ship Stabilization Systems. ONR and Bur. Ships Contract N6-ONr-25129, Dept. Electrical Eng., Stanford Univ., Feb. 1953.

30. Spitzer, Lyman, Jr.: Perturbations of a Satellite Orbit. Jour. British Interplanetary Soc., vol. 9, no. 3, May 1950, pp. 131-136.

31. Buchheim, R. W.: Lunar Instrument Carrier - Attitude Stabilization. U.S. Air Force Project RAND Res. Memo. RM-1730 (ASTIA Doc. No. AD 112402), The RAND Corp., June 4, 1956.

TABLE I.- SPACE ENVIRONMENTAL MISSIONS

Item	Instrumentation		Orbit	Duration (a)	Stabilization control (b)
	Instrument	Weight, lb			
Physics					
Atmospheric:					
Composition	Mass spectrometers	100	High inclination, moderate ellipticity, 125-mile perigee	[Short]	[Coarse]
Total density	Large sphere accelerometer	20	High inclination, 125-mile perigee	[Short]	[No spin]
Electron and ion density	Langmuir probe	100	High inclination, moderate ellipticity, 125-mile perigee	[Short]	[Coarse]
Air glow or aurorae	2 photocells	20	High inclination, moderate ellipticity, 125-mile perigee or circular, 300 miles	[Short]	[Coarse or determine attitude]
Magnetic fields	Magnetometers	100	High inclination, high ellipticity, low to high altitudes	[Medium]	[Coarse or determine attitude]
Micrometeorites	Erosion gages, etc.	75	High inclination, low ellipticity, above 300 miles	[Medium]	[Coarse or determine attitude]
Cosmic radiations:					
Composition, direction, and energy	Counters	180	High inclination, low ellipticity, 300 miles	[Medium]	[Coarse]
			Equatorial, 22,000 miles	[Long]	
			High inclination, 500-mile perigee, apogee 20,000	[Long]	
Primary electrons and protons	Ion chamber	100	High inclination, moderate ellipticity, high altitude	[Long]	[No spin or coarse]
Primary gamma (X-rays)	-------------------	300	Low inclination, low ellipticity, 200 miles	[Medium]	[Coarse]
Ultraviolet and infrared	-------------------	25 to 50	Low inclination	[Medium]	[Coarse]
Light, color, and intensity	-------------------	15	High inclination, circular	[Medium]	[Coarse]
Galactic noise, below ionospheric cutoff	Directional antenna	50	Perigee > 300 miles	[Long]	[Coarse]
Space environment					
Exposure of:					
Materials	(c)	--------	(c)	(c)	(c)
Equipment	(c)	--------	(c)	(c)	(c)
Living things[d]:					
Bacteria and yeast organisms		10 to 15	In regions of cosmic radiation	[Short]	[No spin]
Flowering plants		100	Avoid X-radiation	[Medium]	[No spin]
Rats		50	Avoid X-radiation	[Short]	[No spin]
Monkeys		200			
Man		> 2,000	Equatorial, 120 miles, circular	[Very short]	[No spin or coarse]

[a]Short duration is defined as less than 3 months and long duration, from 6 to 12 months.

[b]Coarse control is defined as an orientation error greater than $\pm 1^{\circ}$ and fine control, an error less than $\pm 1^{\circ}$.

[c]Instruments, orbits, duration, and stabilization will be similar to those for "Physics" with additional monitoring instrumentation.

[d]Recovery is necessary.

L-431

TABLE II.- ASTRONOMICAL AND ASTROPHYSICAL MISSIONS

Item	Instrumentation		Orbit	Duration (a)	Stabilization control (b)
	Instrument	Weight, lb			
Low-resolution optical scanning in ultraviolet and infrared of the sun, moon, planets, and galaxy (stars, clouds, space)	Telescope with 1° and 5° of resolution	25 to 50	83° retrograde for solar, low inclination for galactic	[Long]	[Coarse]
High-resolution optical survey in far ultraviolet of the sky	Telescope with 20 seconds of resolution	300 to 500	High ellipticity, high altitude	[Very long]	[Fine]
Solar radiation		300	High inclination	[Very long]	[Fine]
Radio astronomy, below ionospheric cutoff, of solar, planetary, galactic (stars) noise	Directional antenna	50	Perigee above 300 miles	[Long]	[Coarse for survey, fine for pointing]

[a]Short duration is defined as less than 3 months and long duration, from 6 to 12 months.
[b]Coarse control is defined as an orientation error greater than $\pm 1^\circ$ and fine control, an error less than $\pm 1^\circ$.

TABLE III.- METEOROLOGICAL MISSIONS

Item	Instrumentation		Orbit	Duration	Stabilization control (a)
	Instrument	Weight, lb			
Cloud cover plus atmospheric heat balance	TV, data storage, and bolometers	250	51° to 83° retrograde, 300-mile circular or higher	[As long as possible]	[Coarse to fine]
Atmospheric heat balance only	Bolometers and data storage	100	51° to 83° retrograde, 300-mile circular or higher	[As long as possible]	[Coarse to fine]

[a]Coarse control is defined as an orientation error greater than $\pm 1^{\circ}$ and fine control, an error less than $\pm 1^{\circ}$.

L 4 3 1

TABLE IV.- COMMUNICATION MISSIONS

Item	Instrumentation		Orbit	Duration (a)	Stabilization control (b)
	Instrument	Weight, lb			
Propagation and ionospheric measurements:					
Electron density	Long wire, resonant element in oscillator	25	High inclination, moderate ellipticity, perigee less than 125 miles	[Short]	[No spin or coarse]
Electron density above F-layer	Single frequency and multiple frequency	50	High inclination, perigee greater than 300 miles	[Medium]	[Attitude determination or coarse]
Magneto-ionic ducts	VLF receiver	35	High inclination, low ellipticity	[Short]	[No spin or coarse]
Propagation	VHF receiver	50	High inclination, moderate ellipticity, perigee less than 125 miles	[Short]	[Attitude determination or coarse]
Satellite communication	[Erectable antennas mounted on satellite]	--	[300 to extreme distances]	[Medium]	[Fine]
Worldwide communication relays:					
Active	[Large erectable parabolic antennas and equipment]	--	1 at 24-hour equatorial or several at 2,000 miles	[As long as possible]	[Fine]
Passive (nonspherical)	Large erectable reflecting structures, plane mirror, [segment of sphere]	--	1 at 24-hour equatorial or several at 2,000 miles	[As long as possible]	[Fine, coarse]

[a]Short duration is defined as less than 3 months and long duration, from 6 to 12 months.

[b]Coarse control is defined as an orientation error greater than $\pm 1^{\circ}$ and fine control, an error less than $\pm 1^{\circ}$.

TABLE V.- TECHNOLOGICAL MISSIONS

Item	Instrumentation		Orbit	Duration	Stabilization control (a)
	Instrument	Weight, lb			
Precision orbital measurement for: Geodetic	Flashing-light beacon	----------	1,000 miles circular	----------	[Coarse, toward earth]
Relativistic effect on orbits, atomic clock		----------	----------	----------	
Improvement of tracking and launch-guidance systems		----------	----------	----------	
Orbit modification and transfer		----------	----------	----------	
Power systems: Solar power collectors (parabolic)	----------	[200 to 300 for 4 kw]	[Above 300 miles]	[As long as possible]	[Fine, toward sun]
Cooling radiators	----------	----------	----------	[As long as possible]	[Coarse, away from sun]
Energetic liquids stored at low temperature and shielded from sun	----------	----------	----------	[Several months]	[Coarse, away from sun]
Stabilization systems for: Small payloads	----------	----------	----------	----------	----------
Medium payloads	----------	----------	----------	----------	----------
Large payloads	----------	----------	----------	----------	----------
Guidance systems	----------	----------	----------	----------	----------
Space flight propulsion	----------	----------	----------	----------	----------
Reentry and recovery	----------	----------	----------	----------	[Damping system, may be needed]
Manned satellite	----------	[Over 2,000]	[Low circular, equatorial]	[1 day minimum]	[Coarse, not > ±5°]

[a]Coarse control is defined as an orientation error greater than ±1° and fine control, an error less than ±1°.

TABLE VI.- WEIGHT AND INERTIA OF SATELLITE VEHICLES

USED IN COMPARISON

Payload weight, lb	Maximum inertia, slug-ft^2	
	Without last booster stage	With last booster stage[a] (b)
200	20	[c]3,200
700	200	[c]3,200
3,000	2,000	3,200

[a]This stage is used to augment stabilization by gravitational gradient.

[b]These values provide a maximum restoring moment of 1 in-oz by gravitational gradient.

[c]Erectable structures of 5 pounds for 200 pounds and 4 pounds for 700 pounds are used.

TABLE VII.- DESIGN CONDITIONS FOR SOME POSSIBLE INITIAL LAUNCH AND TRANSIENT CONDITIONS FOR GRAVITATIONAL-GRADIENT ACTIVE-DAMPING SYSTEMS $\left[t_{1/10} = 6 \text{ hours}\right]$

Initial conditions			W_S = 200, 700, and 3,000 lb; I = 3,200 slug-ft^2		
Launch error	Transient at 500 miles				
$\dot{\theta}_o$, $\frac{\text{radians}}{\text{sec}}$	$\dot{\theta}_o$, $\frac{\text{radians}}{\text{sec}}$	θ_{max}, deg	$I\dot{\theta}_o$, $\frac{\text{slug-ft}^2}{\text{sec}}$	Damping torque, in-oz	Damping power, watts
0.0005 (good)	0.0010	35	3.2	3.9×10^{-3}	1.64×10^{-9}
.0010 (fair)	.0020	≈85 (maximum capability)	6.4	7.8	6.56
.0050 (sloppy)	.0100	Vehicle tumbles	32.0	----------	-----------

L 4 3 1

TABLE VIII.- DESIGN CONDITIONS FOR SOME POSSIBLE INITIAL LAUNCH AND TRANSIENT CONDITIONS

FOR RADIATION-SENSING SYSTEMS

Initial conditions			W = 200 lb, I = 20 slug-ft^2			W = 700 lb, I = 200 slug-ft^2			W = 3,000 lb, I = 2,000 slug-ft^2		
Launch error	Transient		$I\dot{\theta}_o$, $\frac{\text{slugs-ft}^2}{\text{sec}}$	Torque, in-oz (1)	Power, watts (1)	$I\dot{\theta}_o$, $\frac{\text{slugs-ft}^2}{\text{sec}}$	Torque, in-oz	Power, watts	$I\dot{\theta}_o$, $\frac{\text{slugs-ft}^2}{\text{sec}}$	Torque, in-oz	Power, watts
$\dot{\theta}_o$, $\frac{\text{radians}}{\text{sec}}$	$\dot{\theta}_o$, $\frac{\text{radians}}{\text{sec}}$	θ_{max}, deg									
0.0005 (good)	0.0010	10	0.020	0.022	7.76×10^{-8}	0.200	0.220	7.76×10^{-7}	2.000	2.200	7.76×10^{-6}
		31.8	.020	.007	2.44	.200	.069	2.44	2.000	.690	2.44
		85	.020	.003	.92	.200	.026	.92	2.000	.257	.92
0.0010 (fair)	0.0020	63.6	0.040	0.014	9.76×10^{-8}	0.400	0.139	9.76×10^{-7}	4.000	1.387	9.76×10^{-6}
		10	.040	.088	62.1	.400	.884	62.1	4.000	8.820	62.1
0.0050 (sloppy)	0.0100	10	0.200	2.200	7.76×10^{-5}	2.000	22.000	7.76×10^{-4}	20.000	220.000	7.76×10^{-3}
		31.8	.200	.690	2.44	2.000	6.900	2.44	20.000	69.000	2.44
		85	.200	.257	.92	2.000	2.570	.92	20.000	25.700	.92

[1]For smaller satellites, perturbation torques are of increasing concern; in particular, equipment on board could increase the control requirement.

TABLE IX.- WEIGHT AND POWER FOR SOME COMPONENTS

OF A STABILIZATION SYSTEM[a]

L
4
3
1

Item	Weight, lb	Power, watts
Thermistor bolometer	0.1	Negligible
Preamplifier[b] for bolometer	.1	5
Power supply unit[c] for bolometer and preamplifier	6	3
Telescope for bolometer	.5	None
Floated rate gyro	1.5	3 (running) ≈5 (heating)
Amplifier for rate gyro[b] and/or servo	.1	5

[a]System weight is independent of satellite weight.

[b]Preamplifier is transistorized.

[c]Unit is needed to stabilize power.

TABLE X.- ESTIMATED WEIGHT AND POWER FOR GRAVITATIONAL-GRADIENT

ACTIVE-DAMPING SYSTEM WITH FLYWHEEL

[Maximum restoring moment, 1 in-oz; $\dot{\theta}_o$ = 0.001 radian/sec]

Item	3-axes separate		2-axes coupled[a] 1-axis separate	
	Weight, lb	Power, watts	Weight, lb	Power, watts
Rate gyros	4.5	9 (running) 15 (heating)	3	6 (running) 10 (heating)
Response network amplifiers	.6 .3	15	.4 .2	10
Flywheel and drive[b]	16	18	16	18
Bracket support	2		2	
Miscellaneous	2	5	2	5
TOTAL	25	62	24	49

[a]This arrangement essentially eliminates 1 rate gyro and 1 flywheel unit and requires 2 units to do work of 3.

[b]For this item a 4-inch wheel, no windage, and 12,000 rpm are assumed, and since it is weightless the same bearings, preload, and power are assumed as for a 10^6 cgs gyro.

TABLE XI.- WEIGHT AND POWER REQUIREMENTS FOR EARTH-RADIATION-SENSING STABILIZATION SYSTEMS

[Launch error is $\theta_o = 0.001$ radian/sec and $\theta_{max} = 63.6^\circ$ for comparison with gravitational-gradient system at its maximum capability]

Item	W_S = 200 lb		W_S = 700 lb		W_S = 3,000 lb	
	Weight, lb	Power, watts	Weight, lb	Power, watts	Weight, lb	Power, watts
Radiation units[a]	3	10	3	10	3	10
Response network	.5	--	.5	--	.5	--
Rate gyros, if used	4.5	9 15	4.5	9 15	4.5	9 15
Amplifiers	.3	15	.3	15	.3	15
Wheel and drive units (if used in lieu of jet controls)[b]	4	2	7	7	24	24
Bracket support	1		1		2	
Miscellaneous	2	3	2	4	3	5
Total less gyros[c]	11	30	14	36	33	54
Total with gyros[d]	15	54	18	60	37	78

[a]Power supply for bolometers charged to payload.
[b]Unit assumed to have 4-inch wheels.
[c]Rate damping signal furnished by proportional-type radiation-sensing signal.
[d]Rate damping signal furnished by rate gyros.

TABLE XII.- ESTIMATED WEIGHT OF VARIOUS SATELLITE STABILIZATION SYSTEMS[a]

System	W_S = 200 lb		W_S = 700 lb		W_S = 3,000 lb	
	Weight, lb	Percent	Weight, lb	Percent	Weight, lb	Percent
Spin, no damping	0	0	0	0	0	0
Spin plus passive damper[b]	4	2	14	2	60	2
Spin plus passive damper plus two horizon-scan units[c]	5	2.5	15	2.1	61	2
Gravity, no damping[d]	[e]5	[e]2.5	[e]4	[e].6	0	0
Gravity plus active damper ($\theta_{max} = \pm 85^\circ$)[d]	29	14.5	28	4	24	.8
Earth scan with gyros (θ_{max} comparable to $\pm 85^\circ$)	15	7.5	18	2.6	37	1.2
Earth or sun tracking, no gyros (θ_{max} comparable to $\pm 85^\circ$)	11	5.5	14	2	33	1.1
Star tracking, no gyros ($\theta_{max} = \pm 10^\circ$)	12	6	16	2.3	41	1.4

[a]The estimated weight is exclusive of weight for power source, and a "fair" injection, $\dot{\theta}_o = 0.001$ radian/sec, is assumed.

[b]Mercury is used.

[c]Power supply for radiation unit is charged to payload.

[d]Maximum restoring moment of 1 in-oz is provided.

[e]Erectable structure is used.

TABLE XIII.- ESTIMATED POWER FOR VARIOUS SATELLITE STABILIZATION SYSTEMS[a]

System	W_S = 200 lb	W_S = 700 lb	W_S = 3,000 lb
	Power, watts	Power, watts	Power, watts
Spin, no damping	0	0	0
Spin plus passive damper	0	0	0
Spin plus passive damper plus two horizon-scan units	5	5	5
Gravity, no damping	0	0	0
Gravity plus active damper $(\theta_{max} \approx \pm 85^\circ)$	49	49	49
Earth scan with gyros $(\theta_{max}$ comparable to $\pm 85^\circ)$	[b]54	[b]60	78
Earth or sun tracking, no gyros $(\theta_{max}$ comparable to $\pm 85^\circ)$	[b]30	[b]36	54
Star tracking, no gyros $(\theta_{max} = \pm 10^\circ)$	[b]33	[b]43	91

[a]A "fair" injection, θ_o = 0.001 radian/sec, is assumed.

[b]The power would be about 6 watts more for a torque capability of 1 in-oz.

TABLE XIV.- RELATIVE EFFORT REQUIRED TO DEVELOP

SATELLITE STABILIZATION SYSTEMS

System	Relative effort for -		
	W_S = 200 lb (a)	W_S = 700 lb (a)	W_S = 3,000 lb (a)
Spin	1	1	1
Spin plus two horizon scanner units	2	2	2
Gravity, no damping	[b]3	[b]3	1
Gravity, active damping	[b]5	[b]5	3
Horizon scan, area scan	4	4	4
Moon, planet, sun, and star tracking	4	4	4

[a]Numbers denote the order of the amount of effort required, the smallest number denoting the least effort.

[b]Development of an erectable structure is necessary.

TABLE XV.- RELATIVE RELIABILITY OF SATELLITE STABILIZATION SYSTEMS[a]

System	Relative reliability (b)
Spin	2
Spin plus two horizon scanner units	3
Gravity, no damping	1
Gravity, active damping	4
Horizon scan, area scan	5
Moon, planet, sun, and star tracking	5

[a]Reliability after proper injection into orbit.

[b]Numbers denote the order of reliability, the smallest number being the most reliable.

L
4
3
1

TABLE XVI.- RELATIVE POINTING ACCURACY OF

SATELLITE STABILIZATION SYSTEMS

System	Accuracy (a)
Spin	Coarse to fine
Spin plus passive damping	Fine
Gravity, no damping	Very coarse
Gravity, active damping	Fine
Horizon scan, area scan	Coarse to fine
Moon, planet, sun, and star tracking	Fine to very fine

[a]Very coarse is defined as an orientation error greater than $\pm 6^{\circ}$; coarse, greater than $\pm 1^{\circ}$; fine, less than or equal to $\pm 1^{\circ}$; very fine, less than or equal to $\pm 1'$.

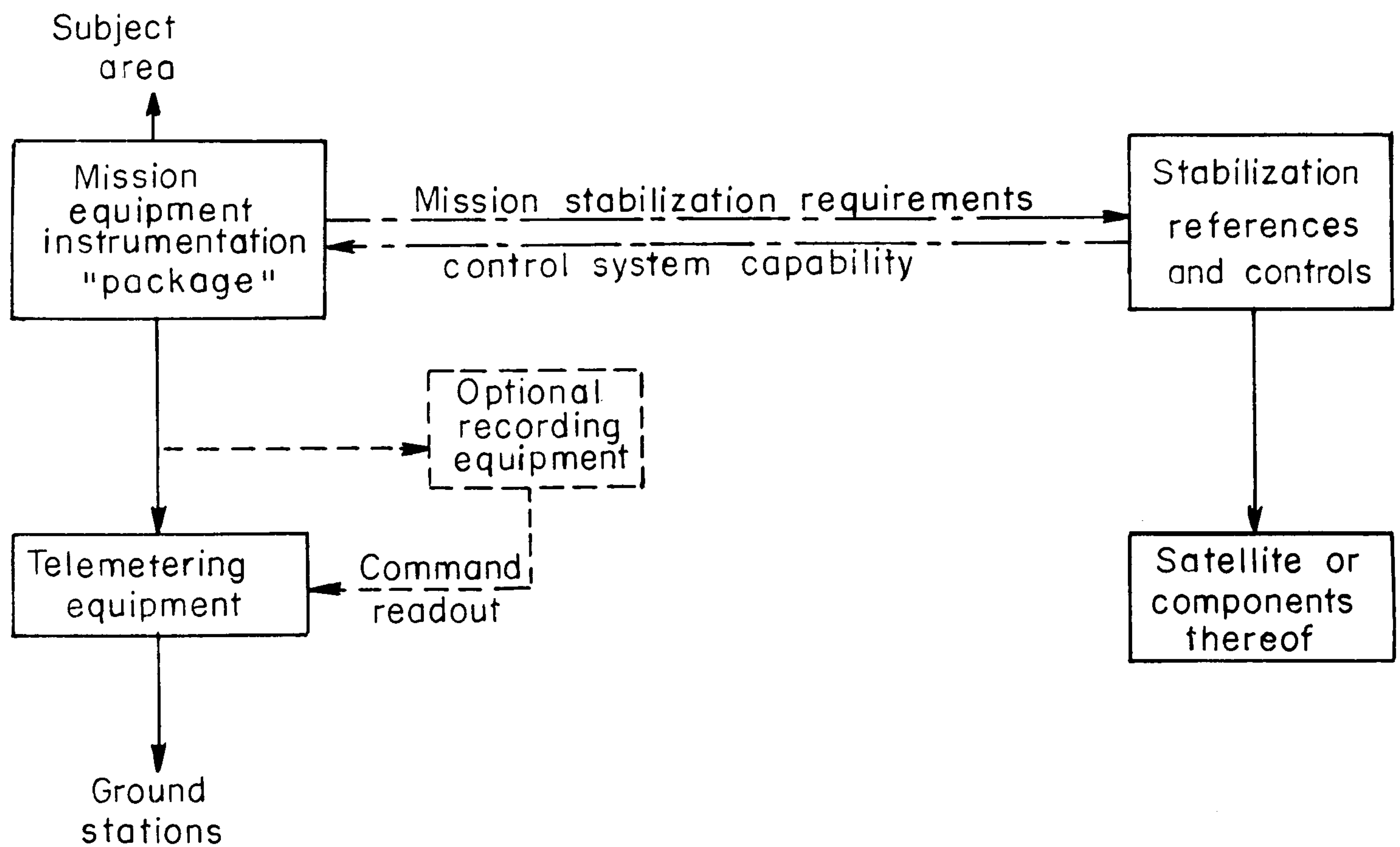


Figure 1.- The problem. Overall system configuration for civilian earth satellite.

L-431

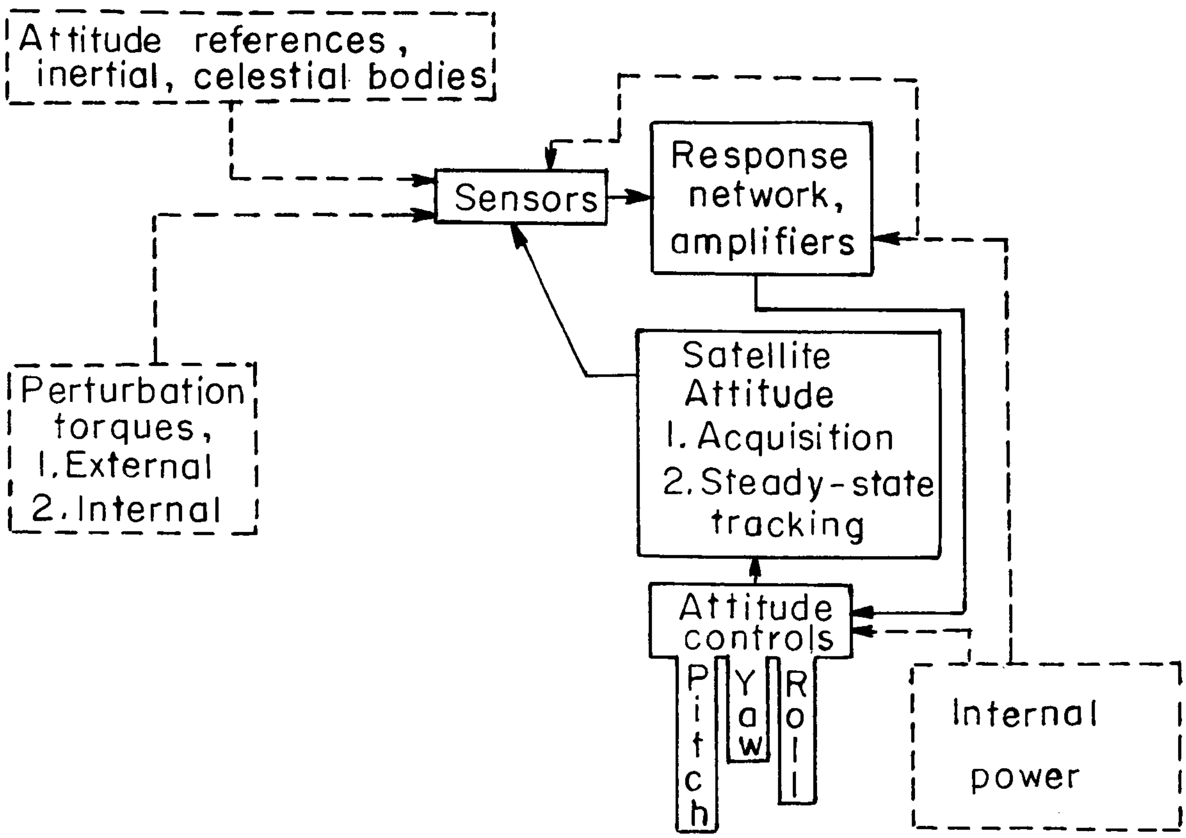


Figure 2.- Information flow diagram of an attitude stabilization system for a satellite vehicle.

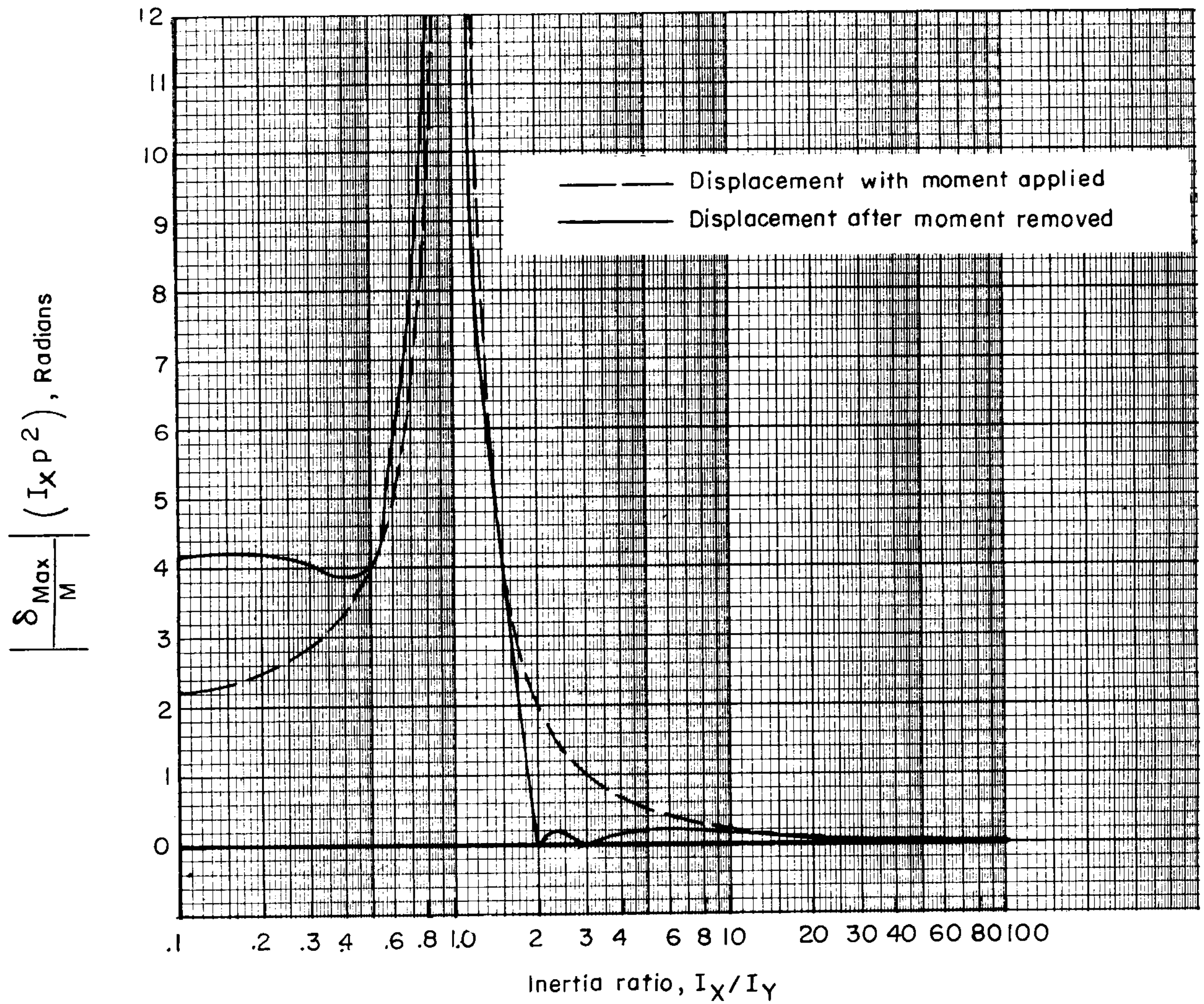


Figure 3.- Variation of maximum displacement parameter with inertia ratio, no damping.

L-431

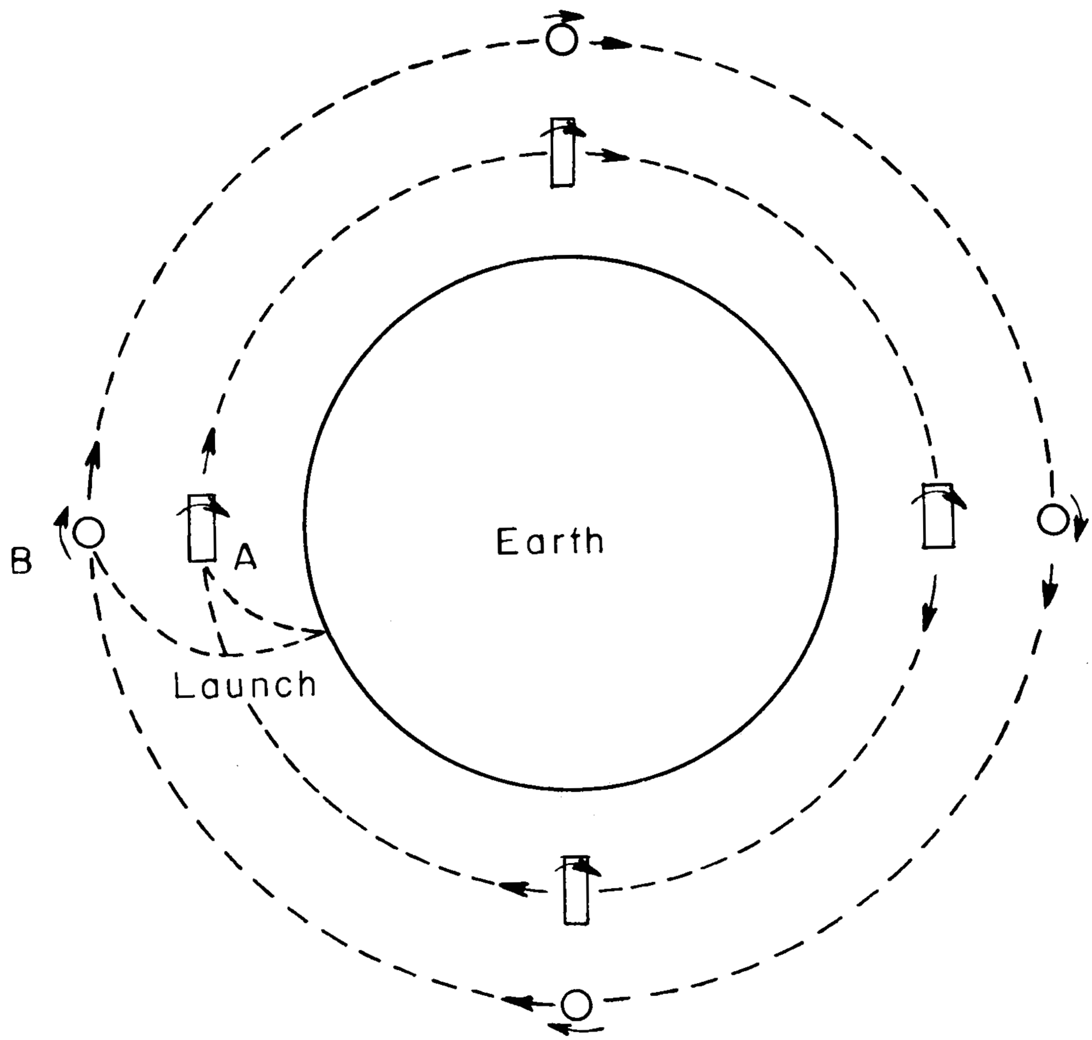
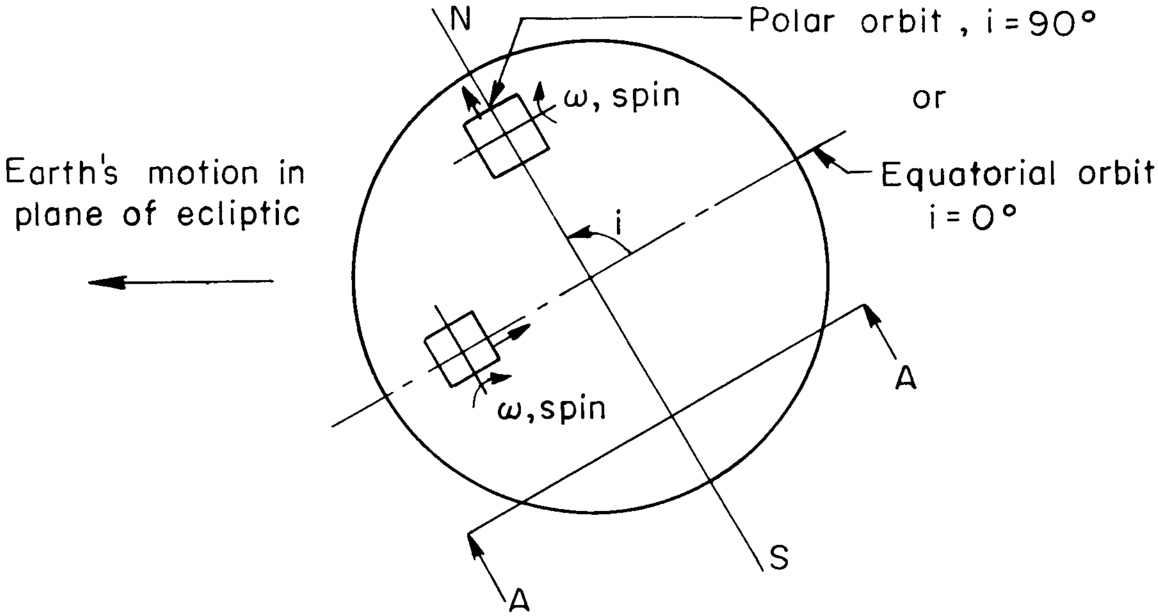
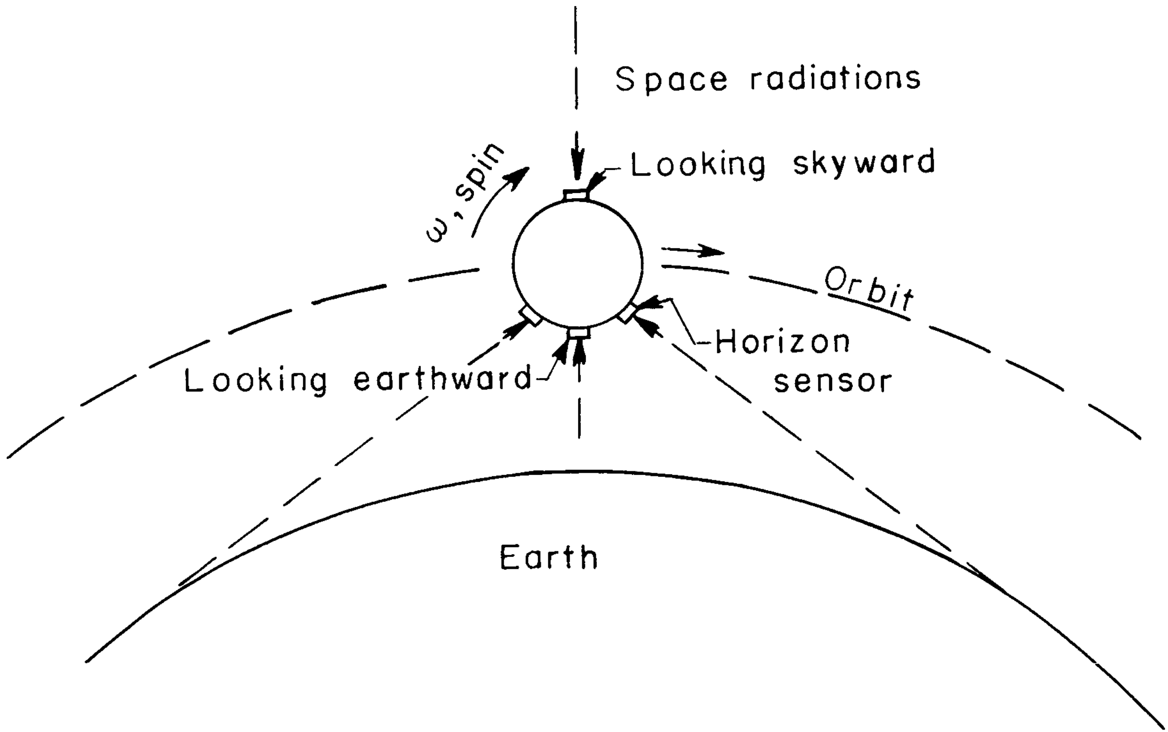


Figure 4.- Two variations of spin stabilization.



(a) Polar and equatorial orbits.

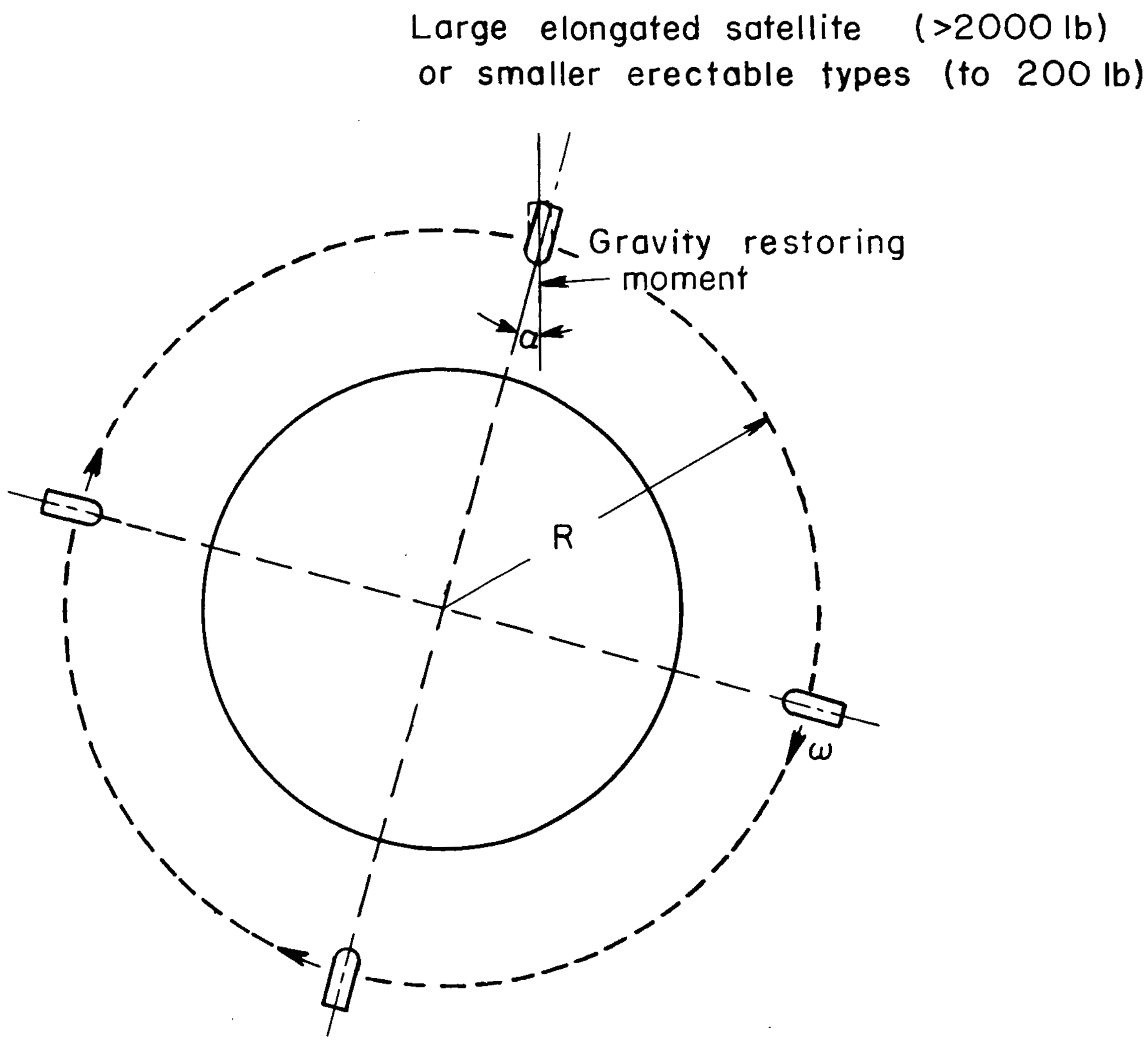


(b) View A-A of equatorial orbit.

Figure 5.- Use of spin in combination with a horizon sensor and polar or equatorial orbits.

L-431

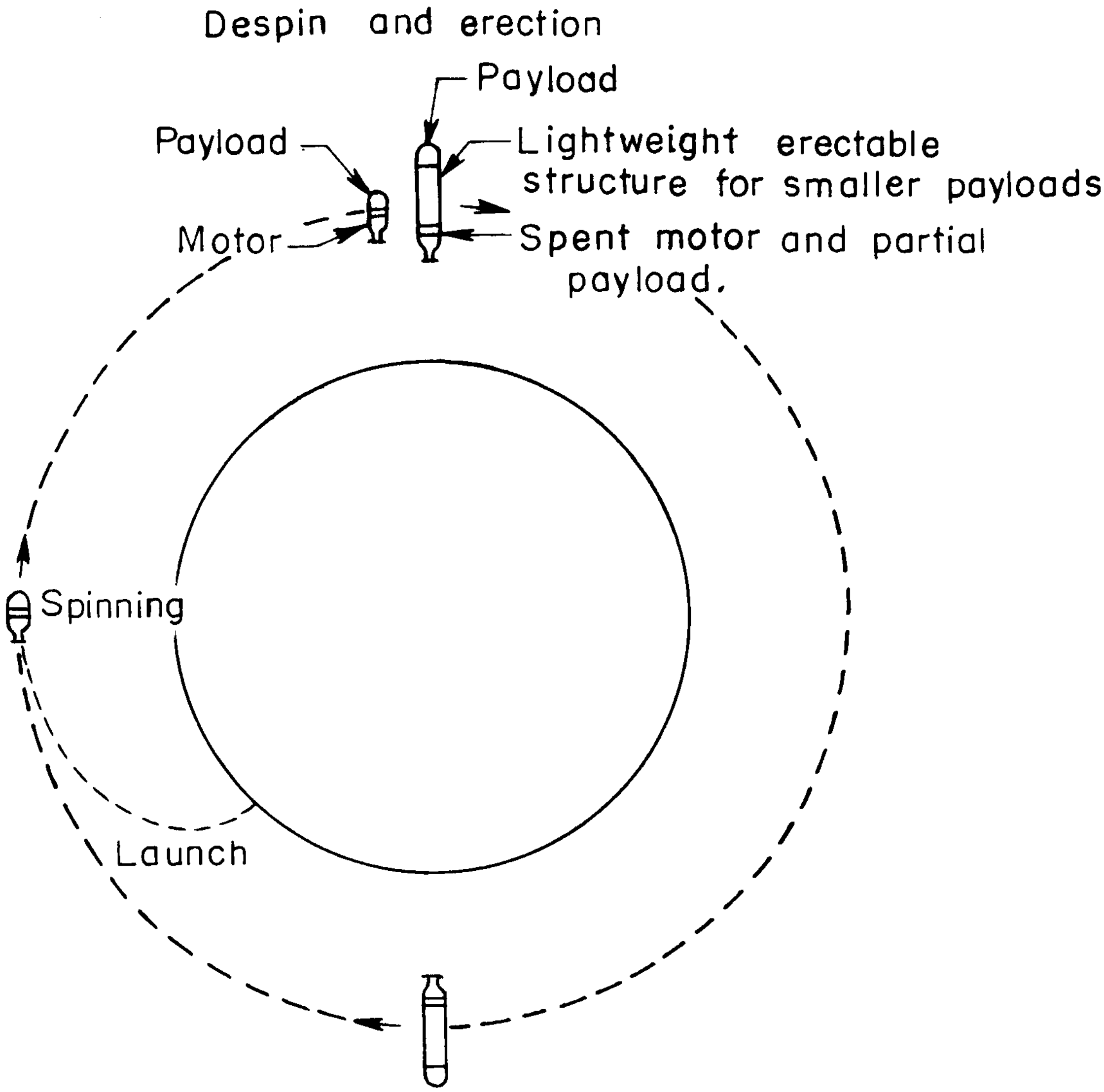
L-431



Restoring moment $= \frac{3\mu}{R^3} \sum_{n=1}^{i} m_n d_n^2 \sin 2\alpha_n$

or for circular orbit, moment $= 3\omega^2 \sum_{n=1}^{i} m_n d_n^2 \sin 2\alpha_n$

Figure 6.- Satellite stabilized by gravitational gradient.

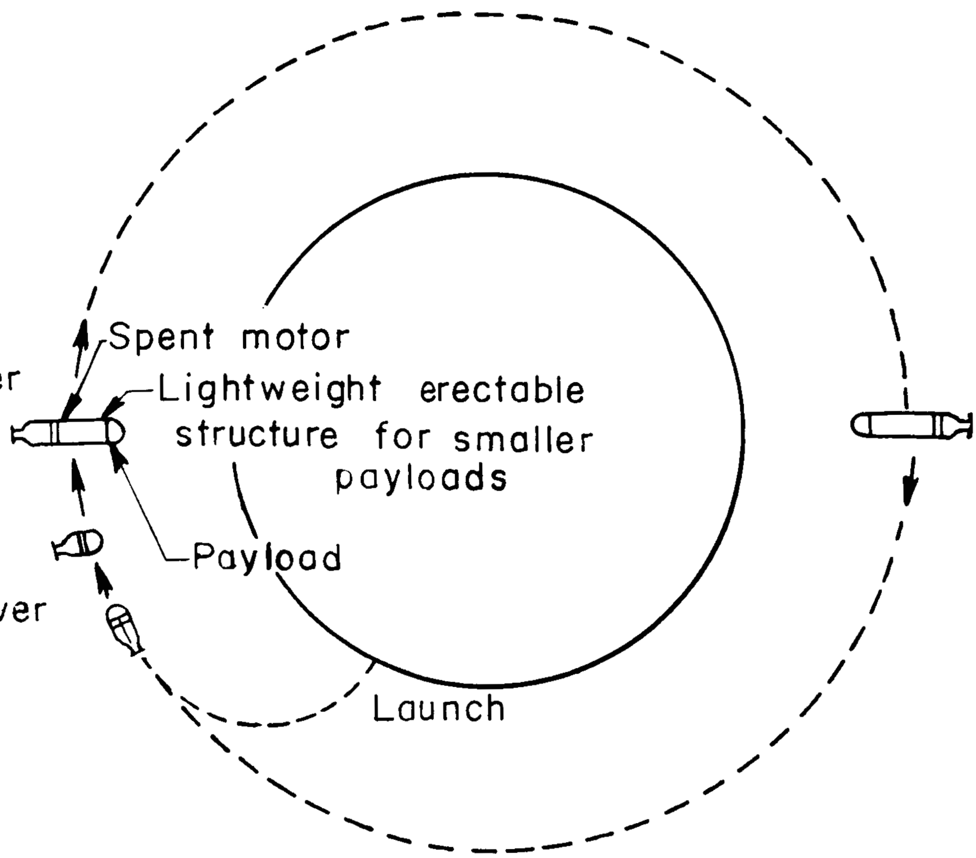


(a) Last rocket stage spin stabilized.

Figure 7.- Two launching schemes for satellites stabilized by gravitational gradient.

L-431

L-431



(b) Last rocket stage nonspinning and programed to pitch 90° down.

Figure 7.- Concluded.

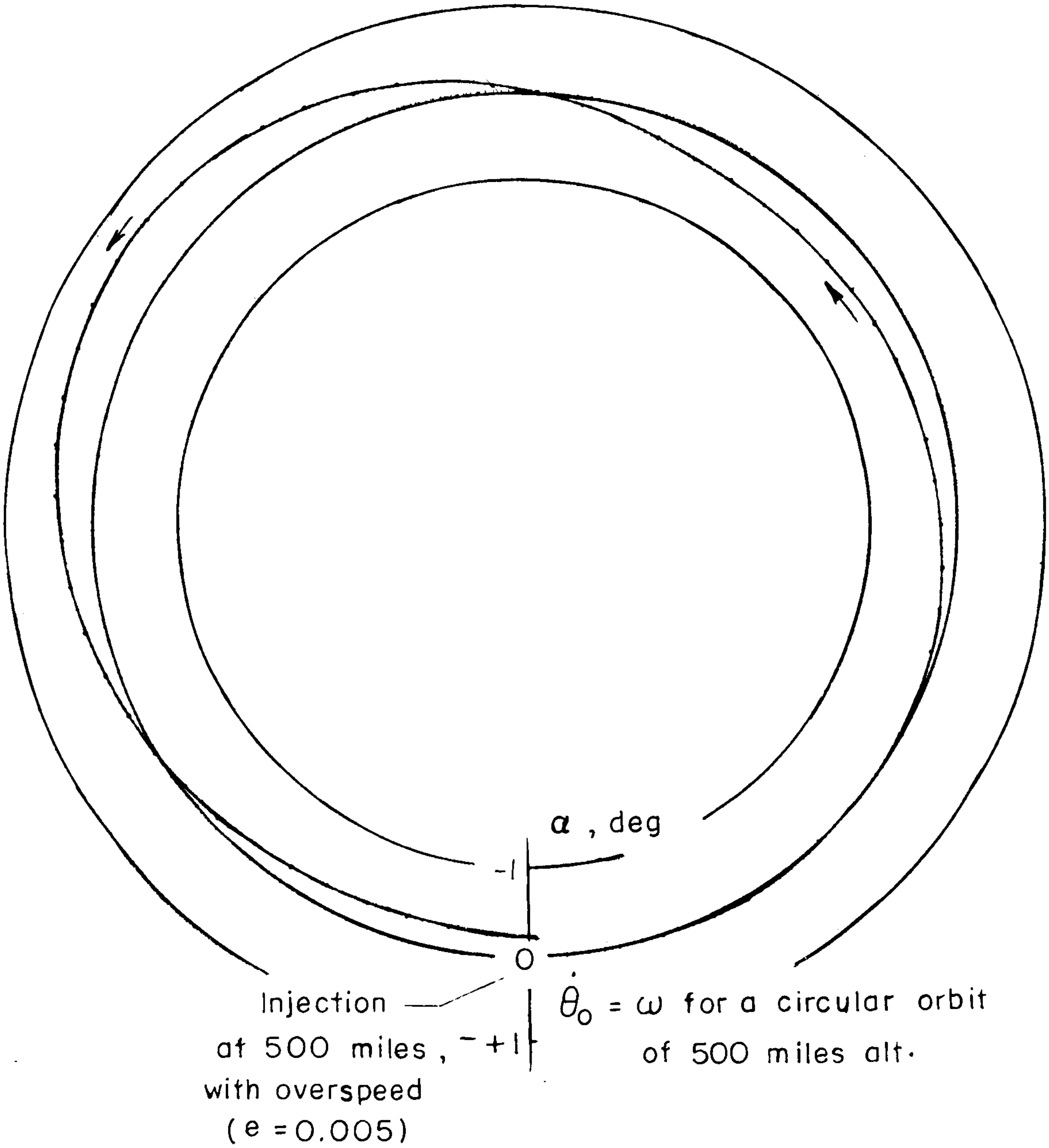


Figure 8.- Effect of an eccentric orbit (e = 0.005) on attitude toward earth.

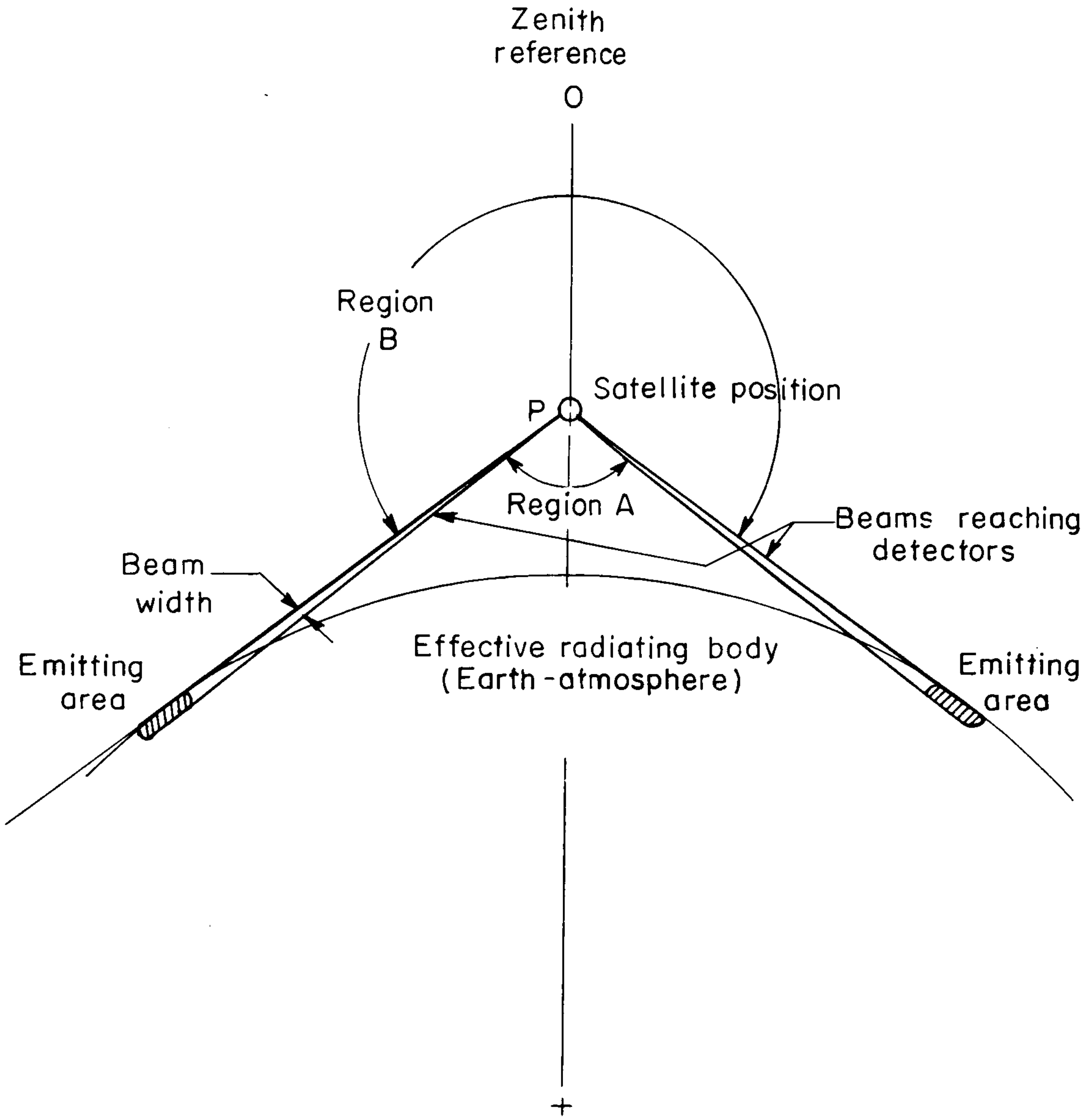


Figure 9.- Earth horizon scan using electromagnetic radiations from earth-atmosphere.

L-451

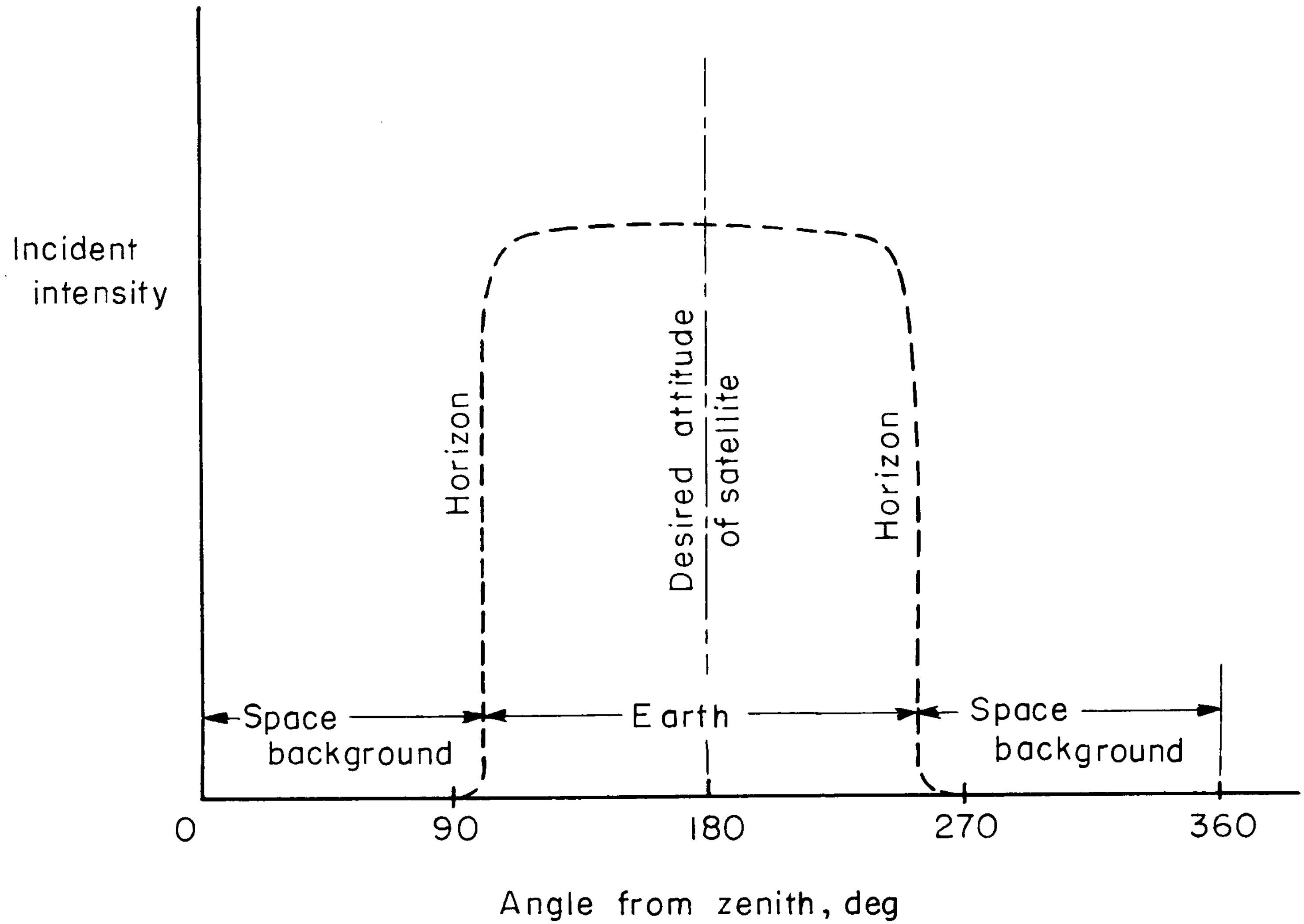


Figure 10.- Typical variation of infrared radiation intensity arriving at satellite (excluding sun, stars, etc. in space background).

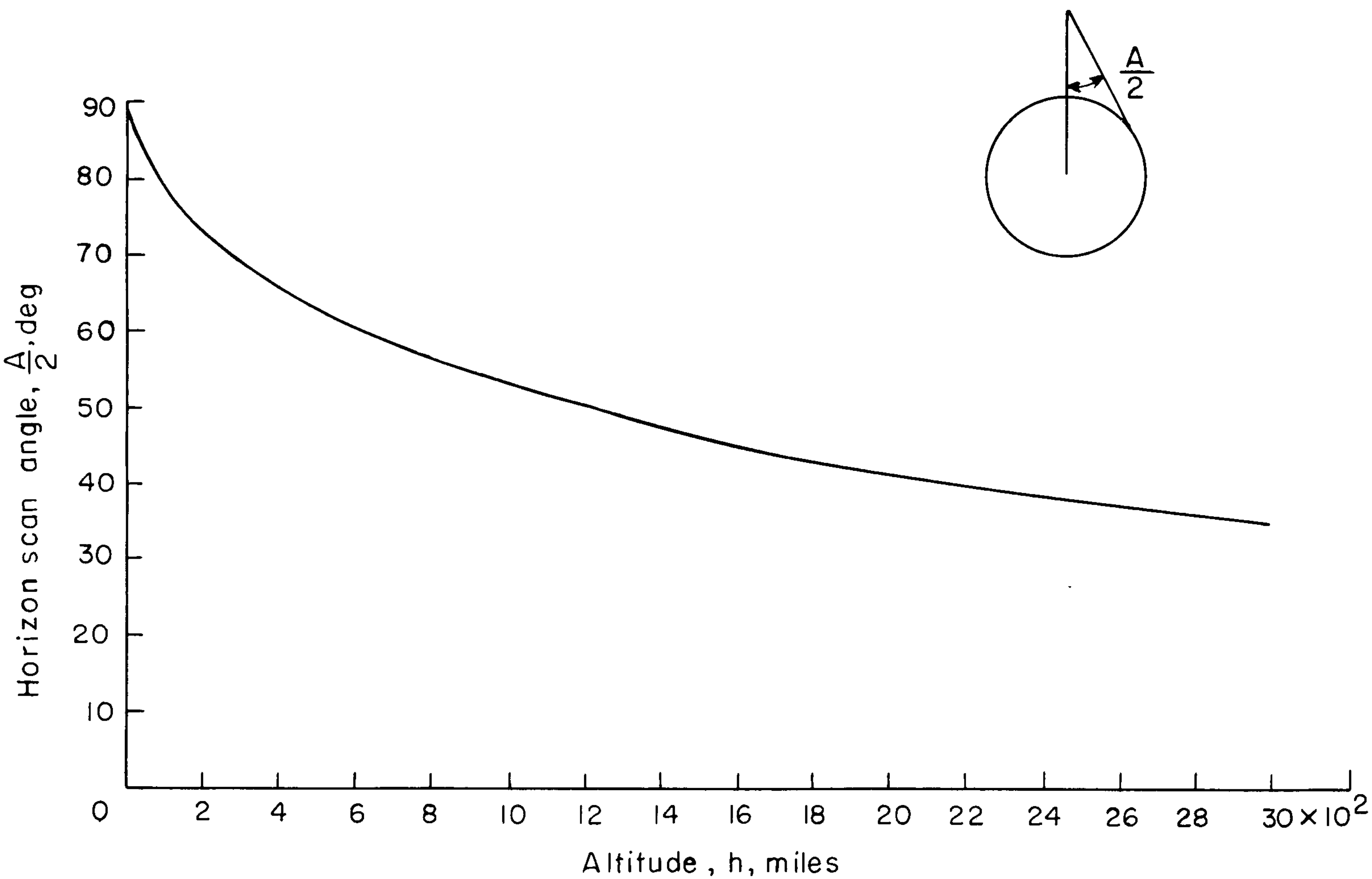


Figure 11.- Variation of horizon scan angle with altitude above earth.

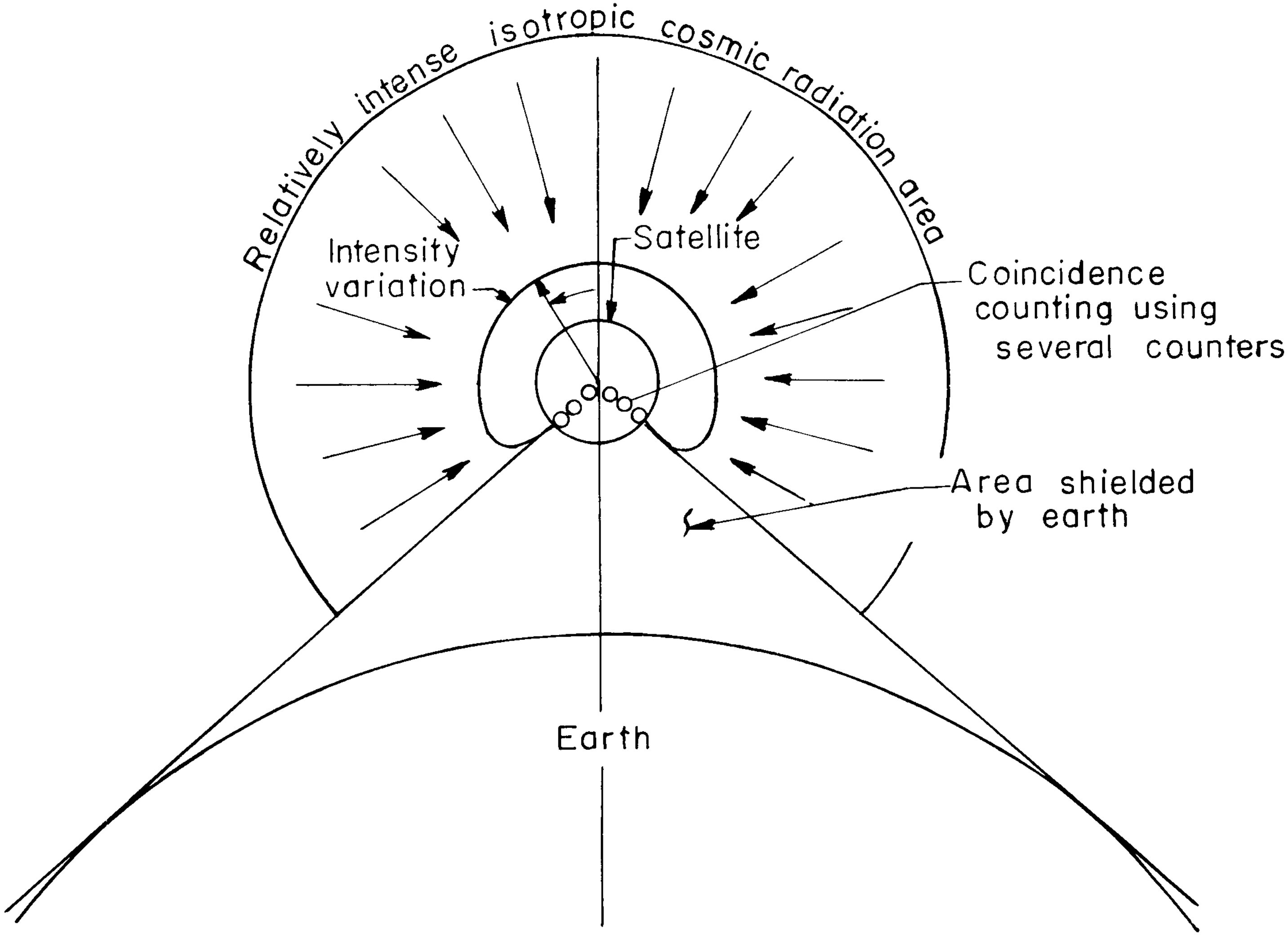
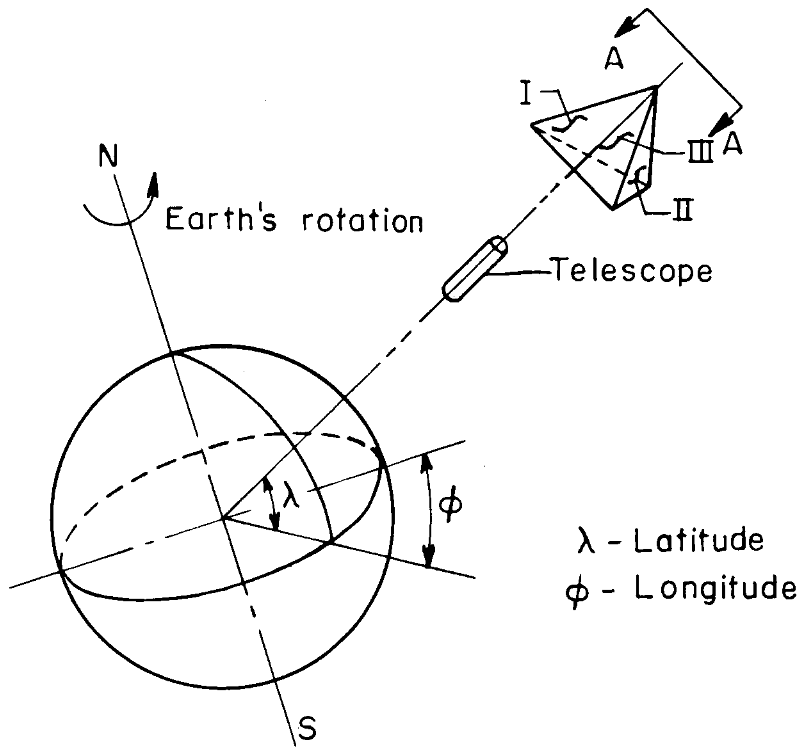
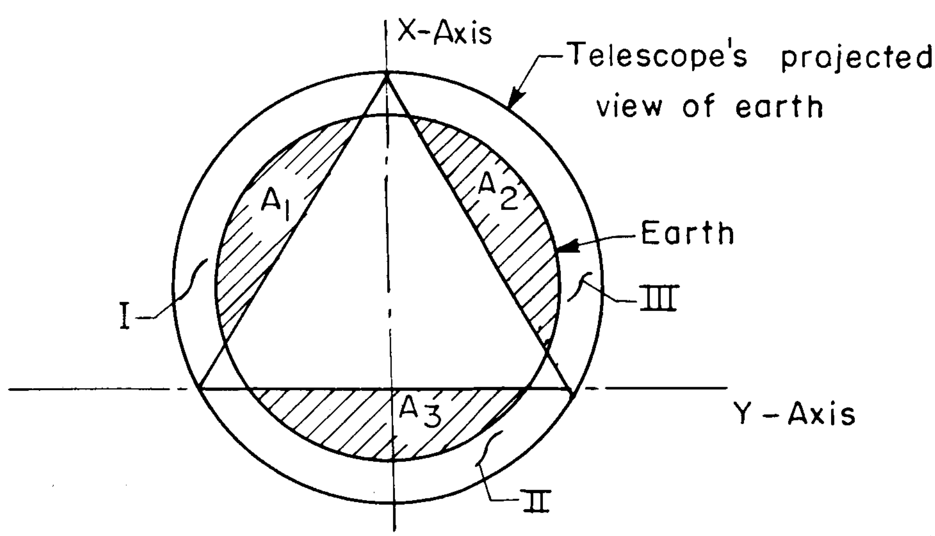


Figure 12.- Horizon scan using Stuhlinger's scheme of cosmic ray shadow effect, reference 19.

L-431



(a) Perspective.



(b) View A-A.

Figure 13.- Earth area scan.

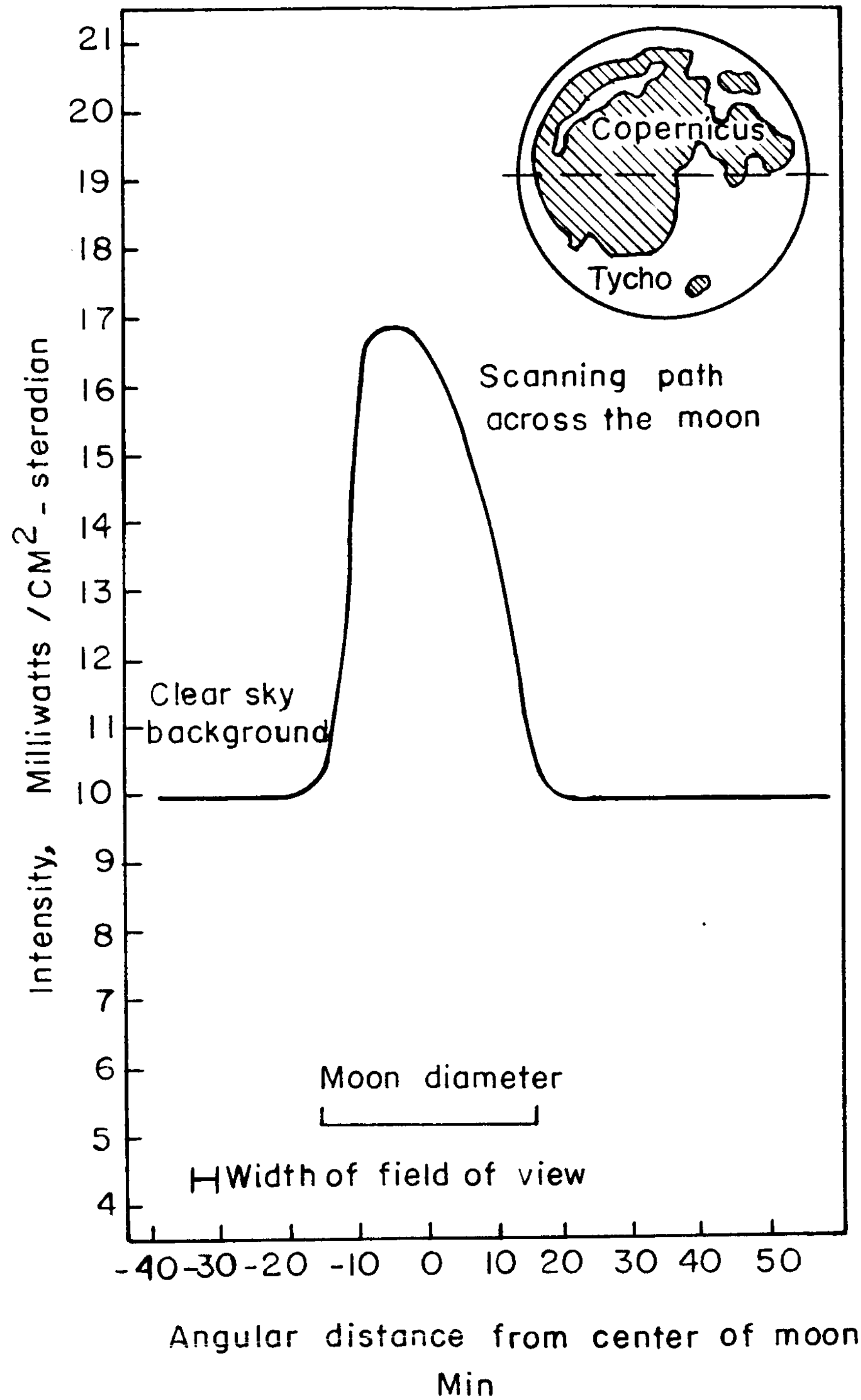


Figure 14.- Measured intensity of thermal radiation from the full moon at 7° elevation angle (from ref. 21).

L-431

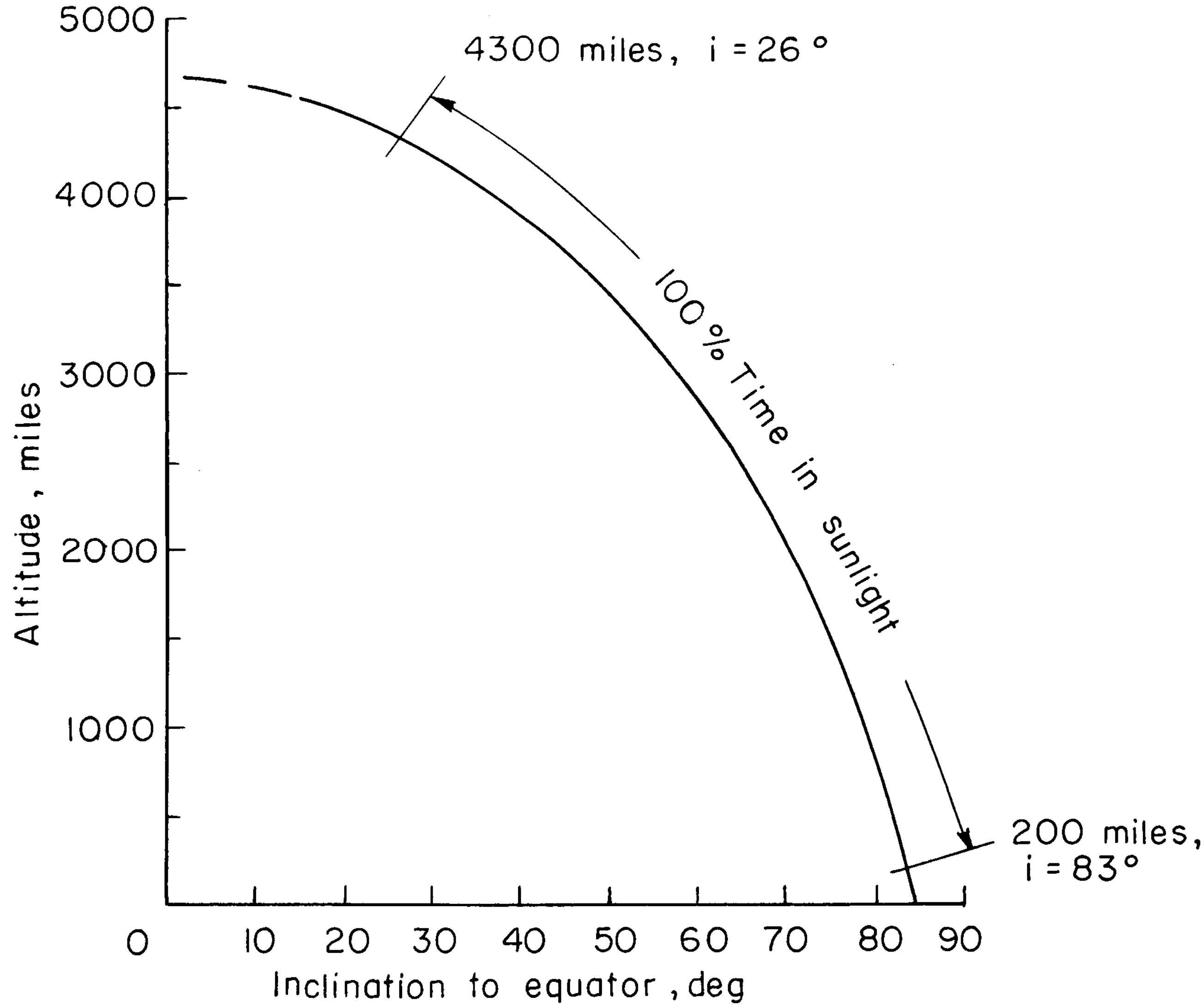
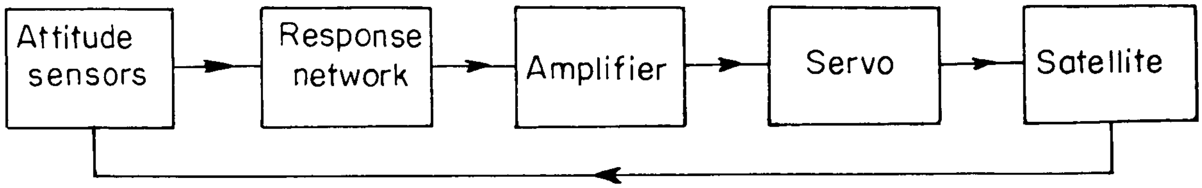
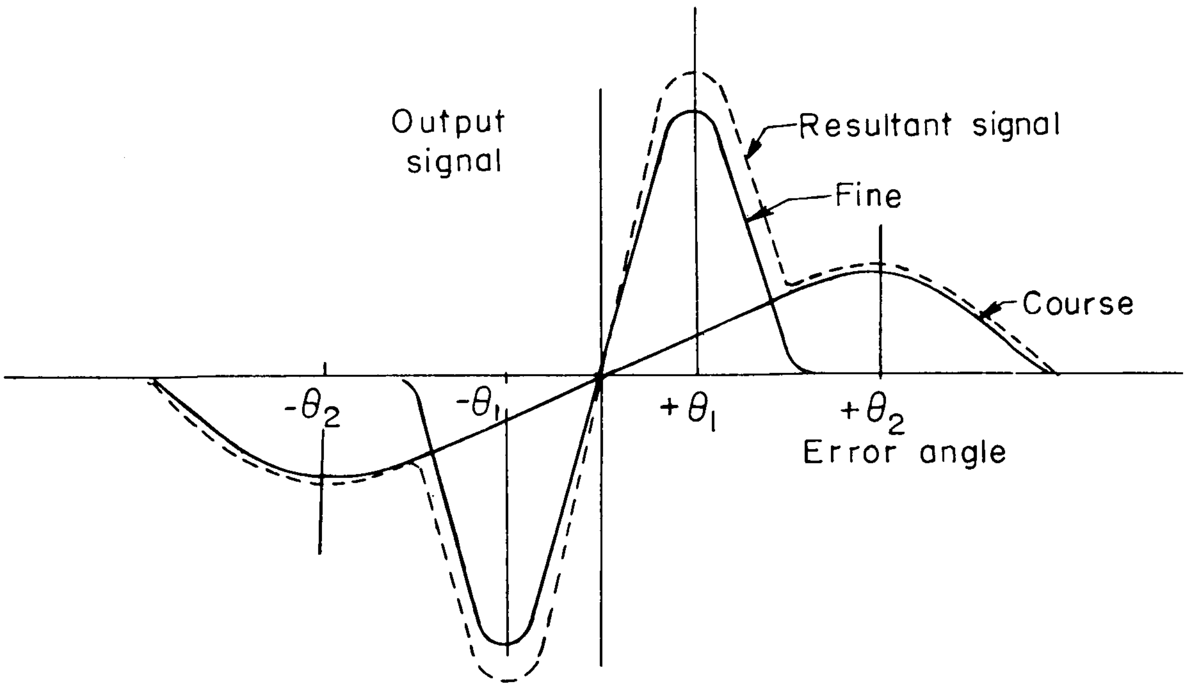


Figure 15.- Circular orbits that permit a continuous view of the sun.



(a) Block diagram of pitch or yaw attitude stabilization.



(b) Attitude sensors output signal.

Figure 16.- Stabilization scheme for astronomical mission, reference 22.

L-431

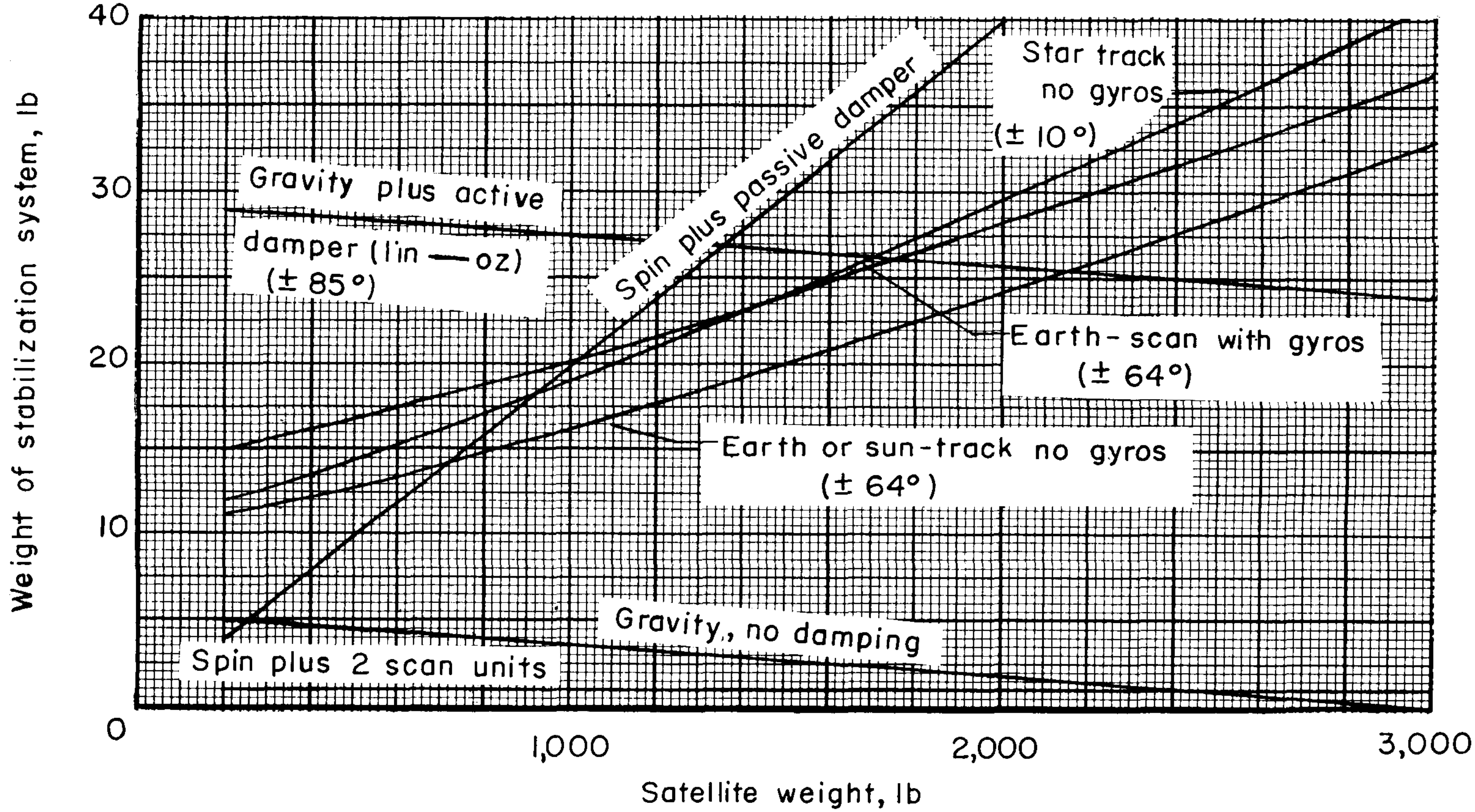


Figure 17.- Weight comparison of stabilization systems based on an initial "capture" maneuver ($\dot{\theta}_{max}$ = 0.001 radian/sec) and use of flywheel control units.

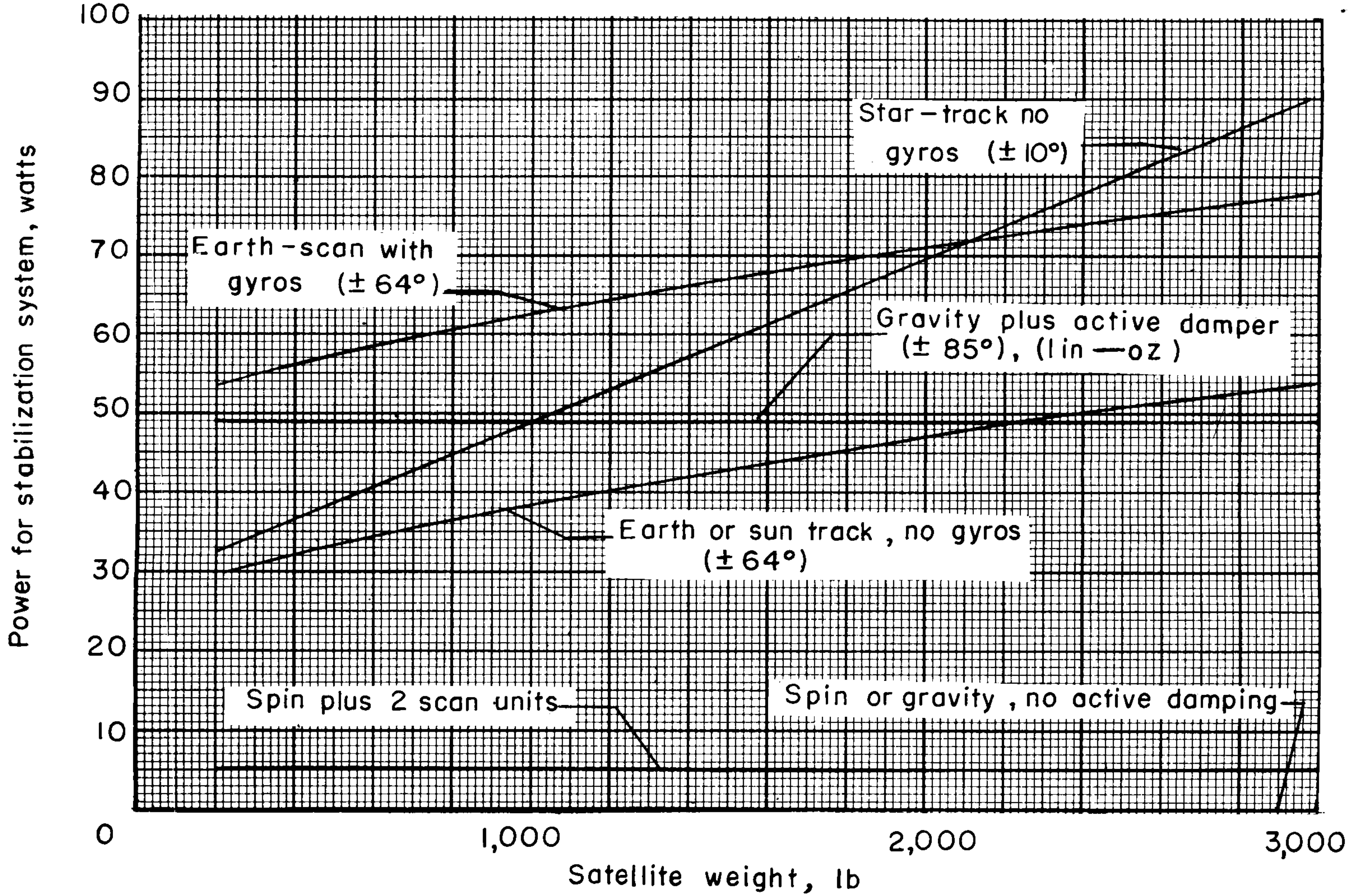
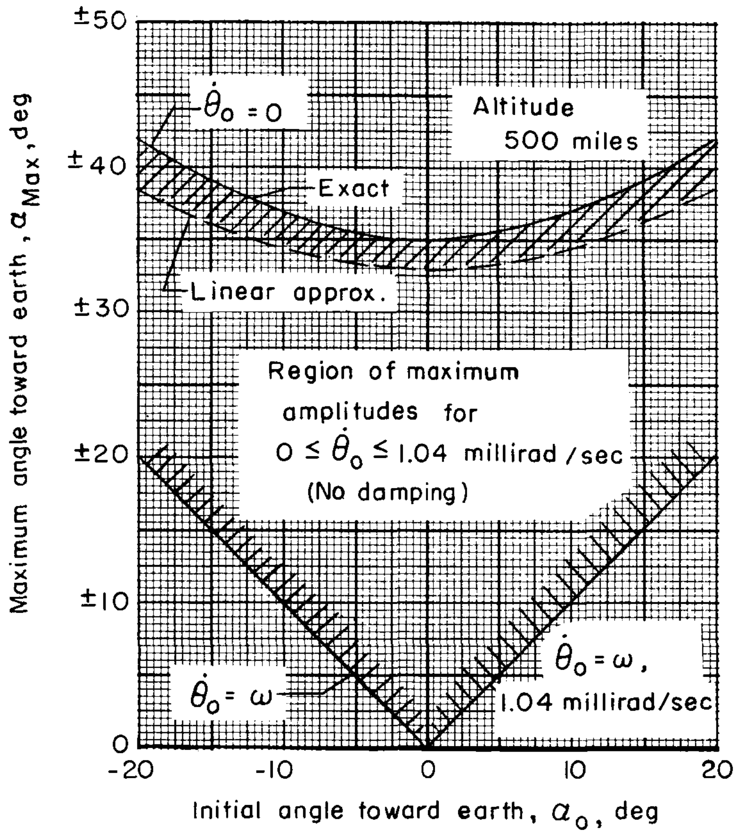
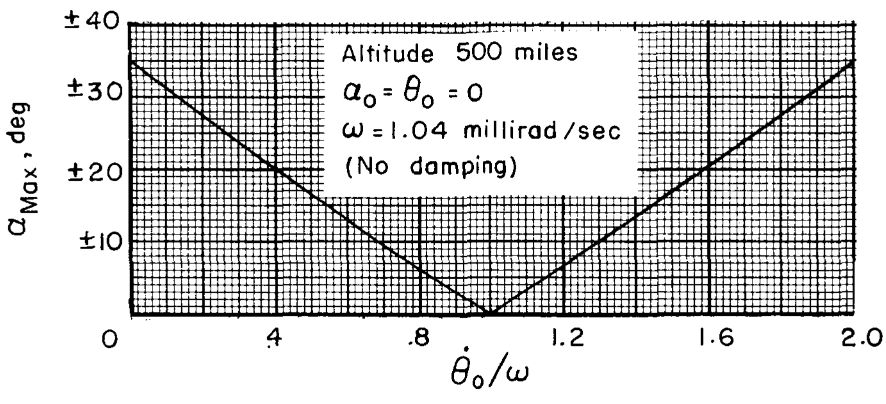


Figure 18.- Power comparison of stabilization systems based on an initial "capture" maneuver ($\dot{\theta}_{max}$ = 0.001 radian/sec) and use of flywheel control units.

L-431

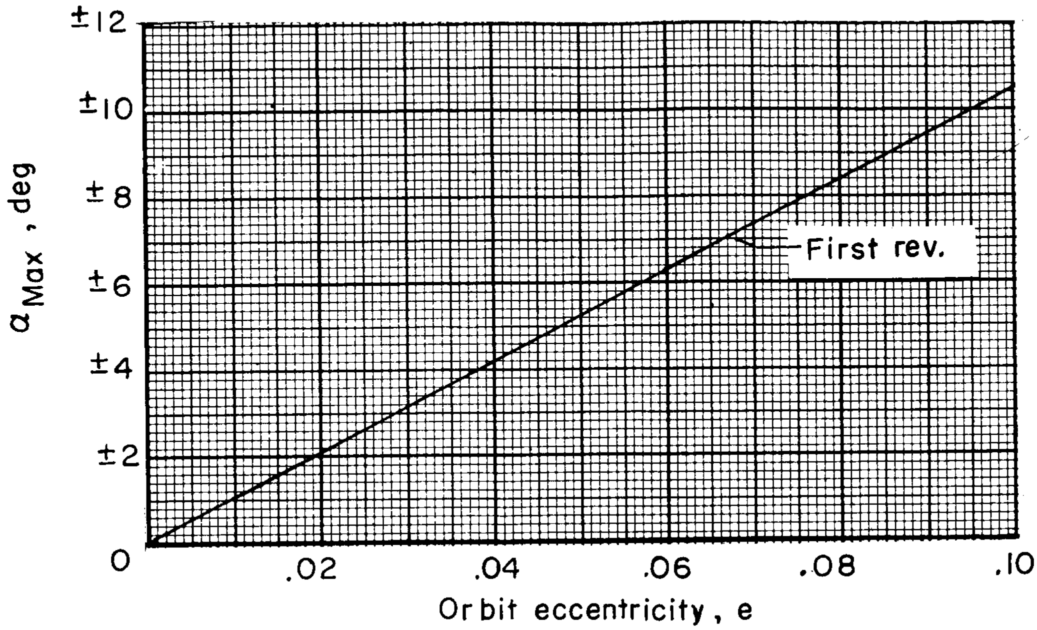


(a) Region of maximum amplitudes, $0 \leqq \dot{\theta}_o \leqq 1.04 \times 10^{-3}$ radian/sec.

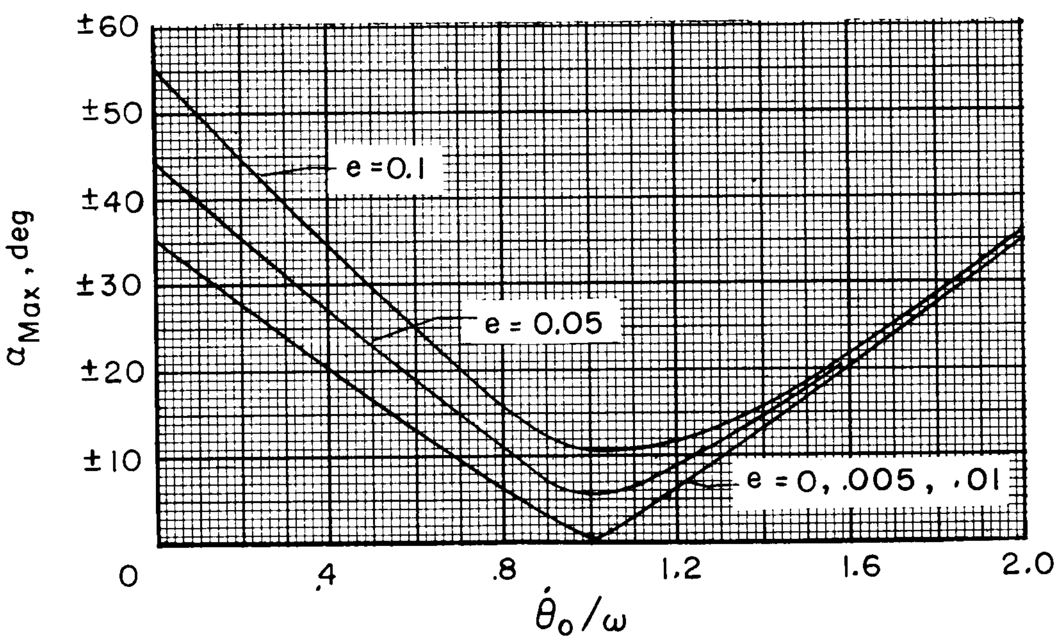


(b) Effect of $\dot{\theta}_o$ on α_{max}.

Figure 19.- Effect of initial conditions on maximum amplitude of pitch orientation toward earth. Circular orbits.



(a) Effect of orbit eccentricity with $\dot{\theta}_o = \omega_{circular}$.



(b) Effect of $\dot{\theta}_o$ for two values of eccentricity, first revolution.

Figure 20.- Amplitude of satellite oscillation with eccentric orbits, 500-mile altitude injection, overspeed condition.

L-431